Design of a Manned Mars Mission using IMPRESS Technology

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Abstract

The objective of this creative investigation is the conceptual design of a propulsion system for a manned mission to Mars. Hydrogen and oxygen were selected as the propellants based upon their historical performance. The propellant storage system was a major consideration of the design due to the problems associated with storing hydrogen and oxygen. The most significant problems are the low density of hydrogen which requires large, heavy storage tanks and cryogenic storage of the propellants.

In order to circumvent these problems, new storage techniques have been incorporated into the design of the propulsion system. Water will be stored in high strength, low density graphite composite tanks until needed. The water will then be electrolyzed into hydrogen and oxygen and cryogenically stored in high strength, low density graphite epoxy composite tanks.

Incorporating these new technologies into the design significantly reduces the problems associated with using hydrogen and oxygen. Using the ideal rocket equation, a top level analysis was performed on five scenarios with different combinations of the presented technologies. Although initial calculations are positive, there are still limitations that must be overcome. Extensive testing and space qualification must be performed before these new technologies will be incorporated into a manned mission. More importantly, a manned spacecraft would be too heavy to launch from the surface of the Earth, therefore requiring that the spacecraft be constructed on orbit.

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Background

In 1957, the Soviet Union launched the world's first satellite Sputnik. This event started an era of intense space investigation, ultimately leading to man landing on the moon. With the continued exploration of the planet Mars a recurring question creeps into ones mind. Will man ever walk on Mars? Considering the curiosity of humans, the answer is most definitely yes. A manned mission to Mars is a complex and sensitive endeavor with two major questions, how and when. The spacecraft that is needed for such a journey would be so large that it would be impossible to launch from the Earth's surface. This leads one to conclude that the spacecraft would have to be constructed in space. Space construction of this magnitude would require a fully operational space station and extensive testing of spacecraft assembled on orbit. Therefore, the question of when such a mission could be accomplished becomes as dubious as how the mission can be accomplished.

Most theoretical manned Mars missions incorporate the use of nuclear technology to provide the necessary thrust levels for such an expedition. Nuclear power in space is an extremely sensitive issue that requires public education and political finesse. An excellent example of public concern about nuclear power in space is the Cassini mission to Saturn. Cassini uses radio-isotropic thermonuclear generators (RTG) which harness energy created by the natural decay of plutonium[1]. An RTG is not a nuclear reactor with moving parts that can fail, but there was still extreme public concern about *nuclear* power on board the spacecraft.

Coupling the political factors surrounding nuclear power in space are the complexity of building a reactor in space and the untested reliability of a nuclear reactor in space.

[1] www.jpl.nasa.gov/cassini/rtg/power.htm

Theoretically a nuclear reactor should work in a space environment, but there have never been any prototypes launched or tests conducted to validate the use of nuclear power in space.

Considering all the negatives surrounding a nuclear powered spacecraft, it seems unlikely to design a mission that uses nuclear technology. Therefore, it is necessary to consider other technology in order to accomplish a manned mission within the next fifty years. A liquid engine that uses hydrogen and oxygen is able to provide a higher specific impulse (Isp) than any other existing technology. Isp is the ratio of the thrust to the weight flow rate of the propellant. In other words, Isp is a measure of the energy content of the propellants and how efficiently it is converted into thrust[2]. The major drawback with using hydrogen and oxygen is long-term cryogenic storage. It would be extremely difficult to cryogenically store the propellants for the six month journey to Mars but impossible to cryogenically store the propellants for two and a half years until the return voyage.

Consider a scenario where the hydrogen and oxygen are stored as water until needed. The water is then separated into hydrogen and oxygen using electrolysis. The use of water eliminates the need for long-term cryogenic propellant storage. Once separated, the propellants will be cryogenically stored in high strength, graphite composite tanks to further reduce the mass of the system. Lawrence Livermore National Laboratory (LLNL) is designing and developing the technological advances that can help make this mission possible.

[2] Larson, W.J. and Wertz, J.R. p. 640.

Introduction

As we approach the twenty-first century, space exploration is continuing at a steady rate. Although NASA is moving toward smaller, less expensive missions, there is still the lingering idea of human exploration of our universe. With a multitude of robot exploration missions of Mars, it is only logical that the next advance will be an astronaut walking on Mars.

At this moment in time, human exploration of Mars is nearly impossible due to the duration of the voyage to and from Mars. In theory the only possible means of travel to Mars would require nuclear power. Political and environmental activists make nuclear travel almost impossible. This leads to new advances in other propulsion systems.

Liquid propulsion is the only other technology available at this time that is capable of producing enough specific impulse to send a manned mission to Mars. Hydrogen and Oxygen are the only propellant combination that a high enough Isp to accomplish such an endeavor. In order to store enough hydrogen and oxygen, cryogenics must be employed. Since the mission to Mars takes well over six months, with at least an eighteen month stay and a six month return journey, it is impossible to cryogenically store the propellants for the return mission.

Although long-term use of cryogenics is not feasible, imagine if water could be stored on board the spacecraft and converted to hydrogen and oxygen as needed. Thanks to Lawrence Livermore National Laboratory that dream is a reality. The Integrated Modular Propulsion and Regenerative Electro-Energy Storage System (IMPRESS) uses solar energy to create hydrogen and oxygen from water by electrolysis.

After the hydrogen and oxygen are created, they will be stored using cryogenics for a short duration until used. Cryogenics must be used in order to liquefy the propellants. Without the cryogenic density, it becomes impossible to store the hydrogen and oxygen. The mass of the storage tanks becomes so great, that the mission is unfeasible. Titanium is presently the strongest, lightest material used for propellant storage tanks. Preliminary testing of graphite/epoxy composite tanks show a strength increase of eight fold compared to similar tanks created from Titanium[3]. Using the composite material further reduces the mass of the storage tanks.

Analysis of the Mars mission begins with a comparison of a Hohmann transfer and a one tangent burn. The transfer orbit was selected by comparing the time of flight with the additional ΔV needed. The ΔV was then used in a top-level analysis of the Mars mission. Different scenarios were considered and analyzed by using different combinations of IMPRESS, titanium tanks, the graphite/epoxy composite tanks and the on board cryogenic storage system. The scenarios were compared and the best one was selected.

[3] Mitlitsky, Groot, Butler, and McElroy. p. 15.

Specific Topics of Review

Mars Transfer Orbit

The transfer orbit is the driving factor in determining the fuel needed. A Hohmann transfer was initially considered because of the efficiency, but the penalty is the time of flight. The Hohmann transfer requires approximately nine months transition time, which is too long for a manned mission. In order to reduce the time of flight, a one tangent burn was considered. By adjusting the angle to the transfer point, time of flight can be optimized versus the ΔV needed. Calculations for Hohmann transfer and One Tangent Burn comparisons are located in Appendix A.

Hohmann Transfer:

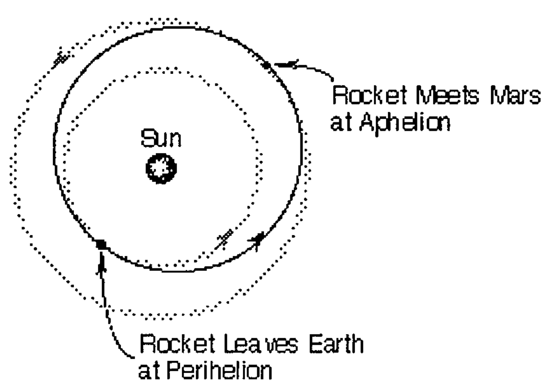

FIGURE 1: SCHEMATIC OF HOHMANN TRANSFER

In order to simplify calculations, the planetary orbits are assumed to be both circular and coplanar[4]. The energy of the transfer ellipse is calculated in order to determine the velocity of the transfer orbit at Earth and Mars.

[4] Bate, R.R, Mueller, D.D., and White, J.E. p. 362.

Equation 1: Energy of Hohmann Transfer Ellipse

$$\varepsilon_T = \frac{-\mu_0}{r_1 + r_2} = \frac{-1}{1+1.524} = -0.3962\frac{AU_{sun}^2}{TU_{sun}^2}$$

ε_T = Energy of Hohmann Transfer Ellipse
μ_0 = Gravitational Parameter of the sun
r_1 = Radius from Sun to Earth in Astronomical Units
r_2 = Radius from Sun to Mars in Astronomical Units

Equation 2: Velocity of Transfer Orbit at Earth/Mars

$$v_1 = \sqrt{2(\frac{\mu_{Sun}}{r_1} + \varepsilon_T)}$$

v_1 = Velocity of Transfer Orbit at Earth/Mars
r = Radius of Earth/Mars Orbit Around Sun

Equation 3: Velocity of Earth/Mars Orbit Around Sun

$$v_{planet} = \sqrt{\frac{\mu_{Sun}}{r_{planet}}}$$

v_{Planet} = Velocity of Earth/Mars Heliocentric Orbit
μ_{Sun} = Gravitational Parameter of Sun (1.3271544 E11 km^3/sec^2)
r_{Planet} = Radius of Earth/Mars Heliocentric Orbit

Equation 4 : Change in Velocity to get on/off Transfer Orbit

$$\Delta V_{on} = V_{tx} - V_{planet}$$
$$\Delta V_{off} = V_{planet} - V_{tx}$$

The velocity of the parking orbits at Earth and Mars are calculated using the gravitational parameters of the respective planet and the altitude of the parking orbit, as indicated in the following equation.

Equation 5: Velocity of Parking Orbit

$$v_{Park} = \sqrt{\frac{\mu_{Planet}}{r_{Park}}}$$

v_{Park} = Velocity of Parking Orbit
μ_{Planet} = Gravitational Parameter of Planet
r_{Park} = Radius of Parking Orbit

Following the patched conic method for interplanetary transfers, the energy of the transfer orbit at each planet is determined using the required change in velocity needed from equation 4 above. The energy at each planet is used to determine the velocity of the Hohmann transfer ellipse at perihelion and aphelion.

Equation 6: Energy of Hohmann Ellipse at perihelion and aphelion

$$\varepsilon = \frac{\Delta v^2{}_{planet}}{2}$$

Equation 7: Velocity of Hohmann Transfer Ellipse at perihelion and aphelion

$$v_h = \sqrt{2\left(\frac{\mu_0}{r_{park}} + \varepsilon\right)}$$

Equation 8: Total Change in Velocity

$$\Delta v_1 = v_h - v_{park}$$

$$\Delta v_2 = v_{park} - v_h$$

$$\Delta v_{Total} = \Delta v_1 + \Delta v_2$$

Considering the affects of extended exposure to the space environment, the time of flight of the transfer orbit becomes the critical factor. The time of flight calculation for a Hohmann transfer from Earth to Mars is shown below.

Equation 9: Time of Flight

$$TOF = \pi\sqrt{\frac{(r_{earth} + r_{mars})^3}{8\mu_{sun}}} = \pi\sqrt{\frac{(2.524)^3}{8}} = 4.4512TU_{sun} = 258.9days$$

TOF = Time of Flight of Hohmann Transfer Orbit

One Tangent Burn:

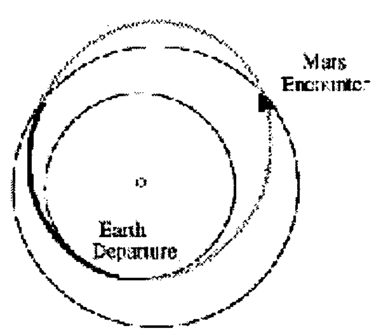

FIGURE 2: SCHEMATIC OF ONE TANGENT BURN

Due to the fact that time of flight is so critical when considering a manned mission, other transfer methods must be employed. The quickest transfer orbit is the one tangent burn

shown above. This transfer orbit consists of a perihelion burn which is tangent to the Earth's orbit about the sun and a non-tangential burn at the intersection with Mars. The angle to the transfer point can be adjusted to optimize the time of flight versus the additional change in velocity. Calculation of a one tangent burn commences with determining the eccentricity and semi-major axis of the transfer orbit as expressed in equations 10 through 12.

Equation 10: Inverse R ratio

$$R^{-1} = \frac{r_i}{r_f} = \frac{1}{1.524} = 0.6562$$

ri = Radius from the Sun to the Earth
rf = Radius from the Sun to Mars

Equation 11: Eccentricity of Transfer Orbit

$$e_{trans} = \frac{R^{-1} - 1}{\cos(\upsilon_{transb}) - R^{-1}} (periapsis)$$

e_{trans} = Eccentricity of Transfer Orbit
v_{trans} = Angle to Transfer Point

Equation 12: Semi-major Axis of Transfer Orbit

$$a_{trans} = \frac{r_{init}}{1 - e_{trans}} (periapsis)$$

a_{trans} = Semi-Major Axis of Transfer Orbit
r_{init} = Radius from the Sun to the Earth

Similar to the Hohmann transfer, the velocity of Earth's orbit, Mar's orbit and the velocity of the transfer orbit had to be calculated in order to determine the change in velocity needed.

Equation 13: Velocity of Earth

$$v_{earth} = \sqrt{\frac{\mu_{sun}}{r_{earth:sun}}}$$

Equation 14: Velocity of Mars

$$v_{mars} = \sqrt{\frac{\mu_{sun}}{r_{mars:sun}}}$$

Equation 15: Velocity of Transfer Orbit at Earth

$$v_{trans_a} = \sqrt{\frac{2\mu}{r_{init}} - \frac{\mu}{a_{trans}}}$$

V_{transa} = Velocity of Transfer Orbit at Earth
μ = Gravitational Parameter of the Sun in Astronomical Units

Equation 16: Velocity of Transfer Orbit at Mars

$$v_{trans_b} = \sqrt{\frac{2\mu}{r_{fin}} - \frac{\mu}{a_{trans}}}$$

V_{transb} = Velocity of Transfer Orbit at Mars

Equations 17 through 20 were used to calculate the ΔV at Earth, at Mars and the total change in velocity needed for the mission.

Equation 17: Change in Velocity at Earth

$$\Delta v_a = v_{trans_a} - v_{earth}$$

Equation 18: Flight Path Angle for Non-tangential Transfer

$$\phi_{trans_b} = \tan^{-1}\left(\frac{e_{trans}\sin(v_{trans_b})}{1 + e_{trans}\cos(v_{trans_b})}\right)$$

ϕ_{transb} = Flight Path Angle for Non-tangential Transfer

Equation 19: Change in Velocity at Mars

$$\Delta v_b = \sqrt{v_{trans_b}^2 + v_{fin}^2 - 2v_{trans_b}v_{fin}\cos(\phi_{trans_b})}$$

Equation 20: Total Change in Velocity for Mission

$$\Delta v_{otb} = |\Delta v_a| + |\Delta v_b|$$

The main reason for using the one tangent burn transfer orbit is to reduce the time of flight of the transfer orbit. The time of flight can be calculated using equations 21 and 22.

Equation 21: Eccentric Anomaly of Transfer Orbit at Transfer Point

$$E = \cos^{-1}\left(\frac{e_{trans} + \cos(v_{trans_b})}{1 + e_{trans}\cos(v_{trans_b})}\right)$$

E = Eccentric Anomaly of Transfer Orbit at Transfer Point

Equation 22: Time of Flight for One Tangent Burn

$$TOF_{trans} = \sqrt{\frac{a_{trans}^3}{\mu}}\left\{2k\pi + (E - e_{trans}\sin(E)) - (E_0 - e_{trans}\sin(E_0))\right\}$$

TOF = Time of Flight of One Tangent Burn Transfer Orbit

Comparison of the transfer angle versus the time of flight and required change in velocity concludes that a φ of 145 degrees produces a time of flight of 193 days. This is a 66 day savings on the time of flight with an increase in ΔV of only 723.5 meters per second. Hence the transfer orbit is much shorter with a very slight ΔV penalty.

Hydrogen/Oxygen Rocket

Based upon proven technology, hydrogen and oxygen produce the highest Isp of any existing technology. There are a few problems with the storage of hydrogen and oxygen, the size of the tanks and long-term cryogenic storage. Due to the low density of hydrogen, the storage tanks are large and heavy which can increases the spacecraft's inert mass. In order to account for the extra weight, more propellant is added, which increases the weight so more propellant is needed. The other problem is long-term cryogenic storage of the liquid oxygen and liquid hydrogen. This is the limiting factor in using oxygen and hydrogen for interplanetary missions. It would be complex to design the cryogenic system to last the six month journey to Mars. Yet it would be impossible to store the propellants for two and a half years for the return trip. Using new technology, discussed below, the disadvantages of using hydrogen and oxygen can be overcome.

IMPRESS

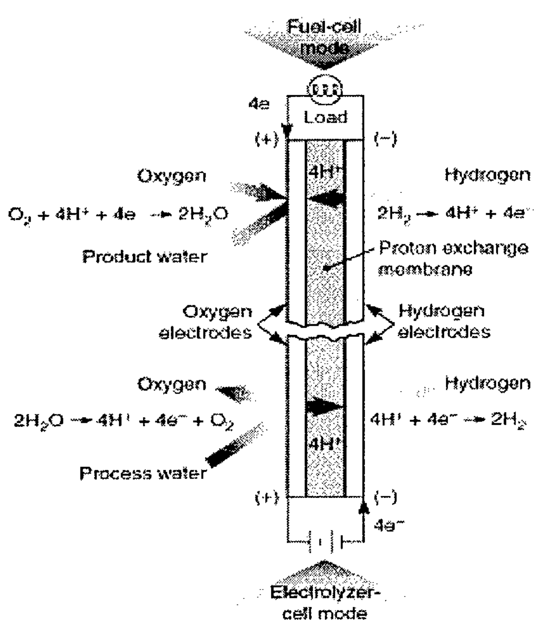

FIGURE 3: IMPRESS

Lawrence Livermore National Laboratory in Livermore, CA has created IMPRESS, which stands for Integrated Modular Propulsion and Regenerative Electro-energy Storage System. IMPRESS is an electrolyzer that uses electrical power to convert water into hydrogen and oxygen. In order to obtain an Isp of 435 seconds an oxidizer to fuel ratio (O/F) of 3.8 is desired, but IMPRESS creates an O/F between 7.5 - 9.1. The extra oxygen created can be used for life support for the manned mission. IMPRESS can also be used to create oxygen for breathing, the hydrogen created can be used for cold gas thruster attitude control. By using IMPRESS, the hydrogen and oxygen are stored on the spacecraft in the form of water eliminating any need for long-term cryogenic storage. After the water is converted into hydrogen and oxygen, cryogenics will be employed to further reduce the mass of the storage tanks.

DOE/Ford T1000G Fuel Bladder

Propellant mass is a major factor when considering the use of hydrogen and oxygen. Due to the low density of hydrogen, the storage tank is very large and heavy. Until recently, titanium was the only low density, high strength material that could be used for storage tanks. Lawrence Livermore National Laboratories has created a graphite composite tank called the T1000G. The T1000G has a proven performance factor of 50,800 meters compared to titanium's performance factor of 6350 meters, this is an eight fold increase in performance. By using T1000G technology, the storage tanks dimensions and mass are greatly reduced, thereby eliminating the other historical problem associated with using hydrogen as a fuel.

On Board Cryogenic Propellant Storage System[5]

Although long-term storage of the propellants in unfeasible, it is necessary to store the hydrogen and oxygen at cryogenic temperature for a short duration before use in order to reduce the size of the storage tanks. Without cryogenics, the densities of the hydrogen and oxygen gas require tanks that are extremely large. This increases the inert mass of the system making the mission impossible. Assuming it is possible to store the hydrogen and oxygen gas at 100 Kelvin without cryogenics, the mass of the storage tanks would be 1.45 million kilograms for the oxygen tank and 6.06 million kilograms for the hydrogen tank. If the spacecraft was outfitted with an 11,000 kilogram cryogenic storage unit[6], the density obtained by using liquid hydrogen and liquid oxygen reduces the mass of the storage tanks to 830 kilograms for the oxygen tanks and 3500 kilograms for the hydrogen tank. Therefore, by adding the cryogenic storage system, the overall system mass is reduced significantly. Analysis was performed using the top-level design described below.

Top Level Design of a Liquid Rocket

After selecting the technology to aid in the mission, a top-level analysis was conducted to determine if this "water" rocket could work. Based upon the selected Isp and an estimated payload mass of 30,000 kilograms and an 11,000 kilogram cryogenic storage system, the ideal rocket equation was used to determine preliminary estimates for propellant, inert structure, initial and final mass.

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[5] Kohout,L.L.
[6] Kohout,L.L. p. 7.

Equation 23: Mass of Propellant (ideal rocket equation)

$$m_{prop} = \frac{m_{pay}\left(e^{\frac{\Delta V}{Isp*g_0}} - 1\right)\left(1 - f_{inert}\right)}{1 - f_{inert} * e^{\frac{\Delta V}{Isp*g_0}}}$$

m_{prop} = mass of propellant
m_{pay} = mass of payload
f_{inert} = inert mass fraction
Isp = Specific Impulse
g_0 = Gravity, 9.81 m/sec2
ΔV = Velocity Change

Equation 24: Mass of Inert Structure

$$m_{inert} = \frac{f_{inert}}{1 - f_{inert}} m_{prop}$$

m_{inert} = Mass of Inert Structure

Equation 25: Mass of Initial Spacecraft

$$m_{init} = m_{pay} + m_{inert} + m_{prop}$$

m_{init} = Mass of Initial Spacecraft

Equation 26: Mass of Final Spacecraft

$$m_{fin} = m_{pay} + m_{inert}$$

m_{fin} = Mass of Final Spacecraft

Before an in depth mission analysis was conducted, equations 23 through 26 were used to create Dummkopf Charts. These charts compare the Isp required versus initial mass for different inert mass fractions based upon payload mass and ΔV required. The Dummkopf charts for this mission are located in Appendix B.

From figure B.12 in appendix B of Humble, Henry and Larson, an oxygen to fuel (O/F) ratio of 3.8 was selected based upon the 435 seconds of Isp. Using figures B.9., B.10, and B.11. and an O/F of 3.8, chamber temperature (Tc), molecular weight (MW) and γ were determined. Engine length, diameter and mass were calculated using the following equations.

Equation 27: Thrust

$$F = \frac{F}{W} * g_0 * M_{init}$$

F = Thrust
F/W = Thrust to Weight Ratio

Equation 28: Mass of Engine

$$m_{engine} = \frac{F}{g_0\left(25.2\log F - 80.7\right)} \text{ kg}$$

m_{engine} = Mass of Engine

Equation 29: Length of Engine

$$L_{engine} = 0.00003042F + 327.7 \text{ cm}$$

L_{engine} = Length of Engine

Equation 30: Diameter of Engine

$$D_{engine} = 0.00002359F + 181.3 \text{ cm}$$

D_{engine} = Diameter of Engine

An expander cycle engine was chosen with regenerative cooling. The expander cycle engine was chosen because of the relative simplicity, low cost and high efficiency. Regenerative cooling will be used in conjunction with the expander cycle. Cooling the thrust-chamber by heat transfer vaporizes the propellants before going into the chamber. In order to keep the system small and simple, the design incorporates a "blow down" concept. A "blow down" system uses extra propellant to maintain pressure in the tanks instead of using pumps or pressurants. This reduces both complexity and weight. Twenty percent extra is added to the propellant mass in order to keep the tanks pressurized.

Cryogenic storage of hydrogen and oxygen provide densities of 71 and 1142 kg/m^3, respectively[7]. These densities do not apply when cryogenics are not being used. In order to determine the density of the hydrogen and oxygen gas at a given temperature and pressure the ideal gas law must be used. The volume of the storage tank was calculated using equations 31 and 32, listed below.

Equation 31: Density of Propellant

$$\rho = \frac{p}{R * Temp}$$

ρ = Density of Propellant
p = Tank Pressure
R = Specific Gas Constant
Temp = Storage Temperature of Propellant

Equation 32: Volume of Propellant

$$V = \frac{m}{\rho}$$

V = Volume of Propellant

[7] Humble, R.W., Henry, G.N., and Larson, W.J. p. 696.

After determining the pressurant requirements, the tank sizes and masses were calculated using the hoop stress and tank factor methods. By using the T1000G graphite composite tank, it was possible to obtain a tank factor (ϕ) of 50,800 meters. This is an eight fold increase over the next strongest material, titanium, with a tank factor of 6350 meters. The mass of the tanks was calculated with the subsequent equations.

Hoop Stress Method

Equation 33: Volume of Cylindrical Tank

$$V_c = \pi r_c^2 l_c$$

V_c = Volume of Cylindrical Tank
r_c = Radius of Cylindrical Tank
l_c = Length of Cylindrical Tank

Equation 34: Surface Area of Cylindrical Tank

$$A_c = 2\pi r_c l_c$$

A_c = Surface Area of Cylindrical Tanks

Equation 35: Thickness of Cylinder Wall

$$t_c = \frac{p_{burst} r_c}{F_{all}}$$

t_c = Thickness of Cylinder Wall
p_{burst} = Burst Pressure
F_{all} = Allowable Material Strength

Equation 36: Mass of Tank - Hoop Stress

$$m_{tank} = A_c t_c \rho_{mat}$$

m_{tank} = Mass of Tank
ρ_{mat} = Density of Tank Material

Tank Factor Method

Equation 37: Mass of Tank - Tank Factor

$$m_{tank} = \frac{p_{burst} * V_{tank}}{g_0 * \phi_{tank}}$$

ϕ_{tank} = Tank Factor

After sizing the tanks, the chamber and nozzle were sized. Columbium is a common material used for chambers and nozzles, therefore it was selected for this design. A bell nozzle was incorporated into the design in order to maximize efficiency. The chamber and nozzle were sized using the ensuing equations.

Equation 38: Exit Area

$$A_e = \frac{A_t}{M_e}\sqrt{\left[\left(\frac{2}{\gamma+1}\right)\left(1+\frac{\gamma-1}{2}M_e^2\right)\right]^{\left(\frac{\gamma+1}{\gamma-1}\right)}}$$

A_e = Exit Area
A_t = Area of the Throat
M_e = Exit Mach Number
γ = Isentropic Parameter

Equation 39: Length of Thrust Chamber

$$L_c = L\frac{A_t}{A_c}$$

L_c = Length of Thrust Chamber (m)
L = Chamber Characteristic Length (m)

Equation 40: Chamber Wall Thickness

$$t_w = f_s p \frac{r_c}{F_{tu}}$$

t_w = Thickness of Chamber Wall
f_s = Factor of Safety
p = Applied Pressure
r_c = Radius of Circular Cylinder
F_{tu} = Ultimate Tensile Strength

Equation 41: Mass of Thrust Chamber

$$m_{tc} = \pi\rho t_w\left(2r_{tc}L_{tc} + \frac{r_{tc}^2 - r_t^2}{\tan\theta_{tc}}\right)$$

m_{tc} = Mass of Thrust Chamber
ρ = Density of Wall Material
r_{tc} = Radius of Thrust Chamber
L_{tc} = Length of Thrust Chamber
r_t = Throat Radius
θ_{tc} = Constant Contraction Half Angle

Equation 42: Exit Diameter

$$D_e = \sqrt{\frac{4\varepsilon A_t}{\pi}}$$

D_e = Exit Diameter
ε = Nozzle Expansion Ratio

Equation 43: Throat Diameter

$$D_t = 2\sqrt{\frac{A_t}{\pi}}$$

D_t = Diameter of Throat

Equation 44: Nozzle Length

$$L_n = \frac{D_e - D_t}{2\tan\theta_{cn}}$$

L_n = Length of Nozzle
θ_{cn} = Nozzle Cone Half Angle

Equation 45: Mass of Nozzle

$$m_n = \pi\rho t_w L_n (r_e + r_t)$$

m_n = Mass of Nozzle
r_e = Nozzle exit Radius
r_t = Throat Radius

After all the mass estimates were complete, a total inert mass and propellant mass was calculated. From this information, an inert mass fraction was determined and compared to the initial selected inert mass fraction. The initial inert mass fraction was adjusted until the two numbers converged.

Discussion of Limitation

The propulsion system of a manned mission to Mars is just one of a multitude of components that must be studied, designed and tested. Similarly, all aspects of an interplanetary manned mission are plagued with problems and limitations. The proposed propulsion system in this design is limited by the new technology presented.

By integrating IMPRESS, the T1000G graphite tanks and an on board cryogenic storage system into the propulsion system many problems associated with a hydrogen/oxygen system are eliminated. The problem is that these are new technologies that have not been space proven as of yet. Due to the complexity and sensitivity of a manned mission, extensive tests need to be conducted to assure that there will be no risk ensued by the astronauts. Although the preliminary ground tests have promising results, they may not pass the rigorous qualification tests in order to prove space worthy for a manned mission.

Aside from the untested technology presented, the mission is also limited by the realization that such a large spacecraft would need to be assembled on orbit. This in itself leads to a complex problem that requires a fully operational space station and extensive testing of on orbit assembly. This factor alone delays any possibility of a manned mission to mars that utilizes this or similar designs.

Discussion of Application

This conceptual design incorporates transfer orbit analysis comparing a Hohmann transfer with a one tangent burn. Using a top-level design, different combinations of the discussed technologies were integrated into the design and compared.

In order to reduce the time of flight, a one tangent burn with a flight path angle of 145 degrees was selected. This reduced the time of flight to 193 days while increasing the ΔV only 723.5 meters per second. For the top-level design, the ΔV is needed for the Earth escape burn and the Mars insertion burn. The Earth escape burn required a ΔV of 3290 meters per second while the Mars insertion burn needed 3067 meters per second of ΔV.

The payload of 30 tonne (30,000 kilograms) and an 11 tonne cryogenic storage system was used to determine the propellant needed for the Mars insertion burn. The propellant required for the Mars burn was added to the payload mass for the Earth escape burn. Using the equations 23 through 26, the inert, initial and final masses were determined for the spacecraft. Because there is no staging, the spacecraft will have a constant inert mass. Realizing this, the inert mass calculated for the Earth escape becomes the inert mass for the Mars insertion calculations. Iteration is necessary until the calculations converge.

Conclusion

Top level analysis of the mission indicates promising results. The use of IMPRESS and the T1000G graphite tanks reduces the overall system mass to workable levels. But, this is only made possible by employing an on board cryogenic storage system. The table below compares the tank masses based upon combinations of the new technologies.

Technology	Mass Hydrogen Tank (kg)	Mass Oxygen Tank (kg)
IMPRESS, T1000G, CSS	3,517	831
IMPRESS, T1000G	6,061,067	1,451,013
IMPRESS, CSS, Titanium	126,094	29,790
CSS, T1000G	5200	1228
CSS, Titanium	13,158,324	3,108,683

FIGURE 4: COMPARISON OF TANK MASS VERSUS TECHNOLOGIES USED

From Figure 4 above it is evident that without the incorporation of this or similar cutting edge technology, that a manned mission to Mars would be improbable using hydrogen and oxygen. Although this design does not conduct a mass breakdown of the specific components of the spacecraft, there is plenty of margin built into the top level design to account for unanticipated extras.

The primary objective of this creative investigation was to illustrate that a manned mission to Mars was possible using existing liquid rocket engine technology. Incorporation of state of the art and next generation storage techniques provide a reduction in the inert mass of the system that make the proposed mission feasible.

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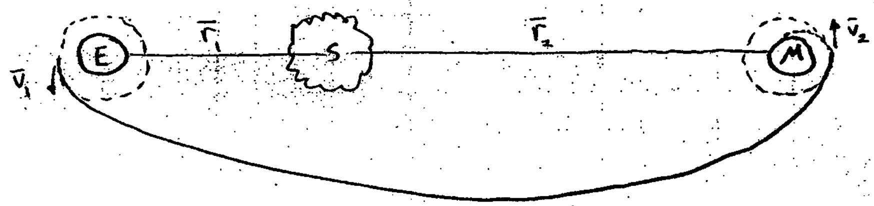
*All equations were extracted from the following sources:

Hohmann Transfer Equations	Bate, Mueller and White
Patched Conic Method	Bate, Mueller and White
One Tangent Burn	Vallado and McClain
Top Level Mission Analysis	Humble, Henry and Larson

Appendix A: Orbit Calculations

Orbit transfer

LEO to Mars Orbit, patched conic Hohmann



$\mathcal{E}_T = -\mu_\odot/(r_1+r_2) = -1/(1+1.524) = -0.3962\ AU_\odot^2/TU_\odot^2$

$V_t = \sqrt{2\left(\frac{\mu_\odot}{r_1}+\mathcal{E}_t\right)} = \sqrt{2\left(\frac{1}{1}+-0.3962\right)} = \sqrt{1.2076} = 1.0989\ AU/TU_\odot$

• vel. of Tx @ Earth

$= 32.73$ Km/sec

T.O.F. $= \pi\sqrt{\frac{(r_\oplus+r_{\text{♂}})^3}{8\mu_\odot}} = \pi\sqrt{\frac{(2.524)^3}{8}} = 4.4512\ TU_\odot = 0.788$ yrs $=$ 258.9 days

$V_\oplus = \sqrt{\frac{\mu_s}{r_{\oplus s}}} = \sqrt{\frac{1.32715E11}{1.496E8}} = 29.78$ Km/s • vel. of earth orbit around sun

$V_{\text{♂}} = \sqrt{\frac{\mu_s}{r_{\text{♂}s}}} = \sqrt{\frac{1.32715E11}{2.278E8}} = 24.14\ \frac{Km}{s}$ • vel. of Mars orbit around sun

$\Delta V_1 = V_{Tx} - V_\oplus = 2.94$ Km/s • ΔV needed @ Earth to get on trans. orbit

$\Delta V_2 = V_{\text{♂}} - \sqrt{\mu_s\left(\frac{2}{r_{\text{♂}}}-\frac{1}{a_t}\right)} = 2.65\ \frac{Km}{s}$ • ΔV needed for Mars insertion

Leaving Earth

$V_\infty = \Delta V_\oplus = 2.94$ Km/s

$V_{park} = \sqrt{\frac{\mu_\oplus}{r_{LEO}}} = \sqrt{\frac{3.986E5}{6878}} = 7.6127$ Km/s

$\mathcal{E} = \frac{V_\infty^2}{2} = \frac{2.94^2}{2} = 4.322$

$V_{h\oplus} = \sqrt{2\left(\frac{\mu_\oplus}{r_{park}}+\mathcal{E}\right)} = \sqrt{2\left(\frac{3.986E5}{6878}+4.322\right)} = 11.1602$ Km/s

$\Delta V_\oplus = 11.1602 - 7.6127 = 3.5475$ Km/s

Entering Mars

$V_{\infty} = \Delta V_{\text{♂}} = 2.65$ Km/s

$V_{park} = \sqrt{\mu_{\text{♂}}/r_{park}} = \sqrt{43053/3893.16} = 3.325$ Km/s

$\mathcal{E} = V_\infty^2/2 = 3.511$

$V_{h\text{♂}} = \sqrt{2\left(\frac{\mu_{\text{♂}}}{r_{park}}+\mathcal{E}\right)} = 5.3979$ Km/s

$\Delta V_{\text{♂}} = 3.3125 - 5.3979 = -2.0854$ Km/s

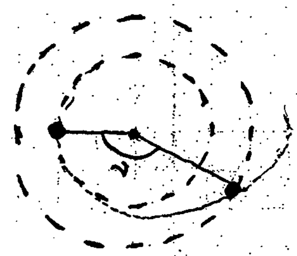
$\boxed{\Delta V_{tot} = 5.6329\ Km/s}$

Orbit transfer

Leo to Mars – one-tangent burn, Patched Conic

$r_1 = 1AU$
$r_2 = 1.524\ AU$

$\nu = 145°$



- $R^{-1} = \frac{r_i}{r_f} = \frac{1}{1.524} = \underline{0.6562}$
- $e_{trans} = \frac{R^{-1} - 1}{\cos(\nu_{trans_i}) + R^{-1}}$ (periapsis)
- $a_{trans} = \frac{r_{init}}{1 + e_{trans}}$ (periapsis)

- Plot shows optimal $\nu_{tx} = 145°$
  $e_{tx} = 0.233029$
  $a_{tx} = 1.30383\ AU$

$\Delta V_\oplus = 0.110418\ AU/TU = 3.28878\ Km/s$
$\Delta V_♂ = 0.145319\ AU/TU = 4.3283\ Km/s$
$\Delta V_{tot} = 0.255737\ AU/TU = 7.61709\ Km/s$

TOF = 192 days

Leaving Earth
$V_\infty = 3.28878\ Km/s$ $\quad V_{park} = \sqrt{\frac{\mu}{r_{park}}} = \sqrt{\frac{3.98665}{6878}} = 7.61268\ Km/s$

$\mathcal{E} = \frac{V_\infty^2}{2} = 5.40804$

$V_{h\oplus} = \sqrt{2\left(\frac{\mu_\oplus}{r_{park}} + \mathcal{E}\right)} = 10.90262\ Km/s$

$\Delta V_\oplus = 10.90262 - 7.61268 = \underline{3.28994\ Km/s}$

entering Mars
$V_\infty = 4.3283\ Km/s$ $\quad V_{park♂} = 3.325\ Km/s$ (Hohmann)

$\mathcal{E} = \frac{V_\infty^2}{2} = 9.36709\ Km/s$

$V_{h♂} = \sqrt{2\left(\frac{\mu_♂}{r_{park}} + \mathcal{E}\right)} = 6.3915\ Km/s$

$\Delta V_♂ = 3.325 - 6.3915 = \underline{-3.0665\ Km/s}$

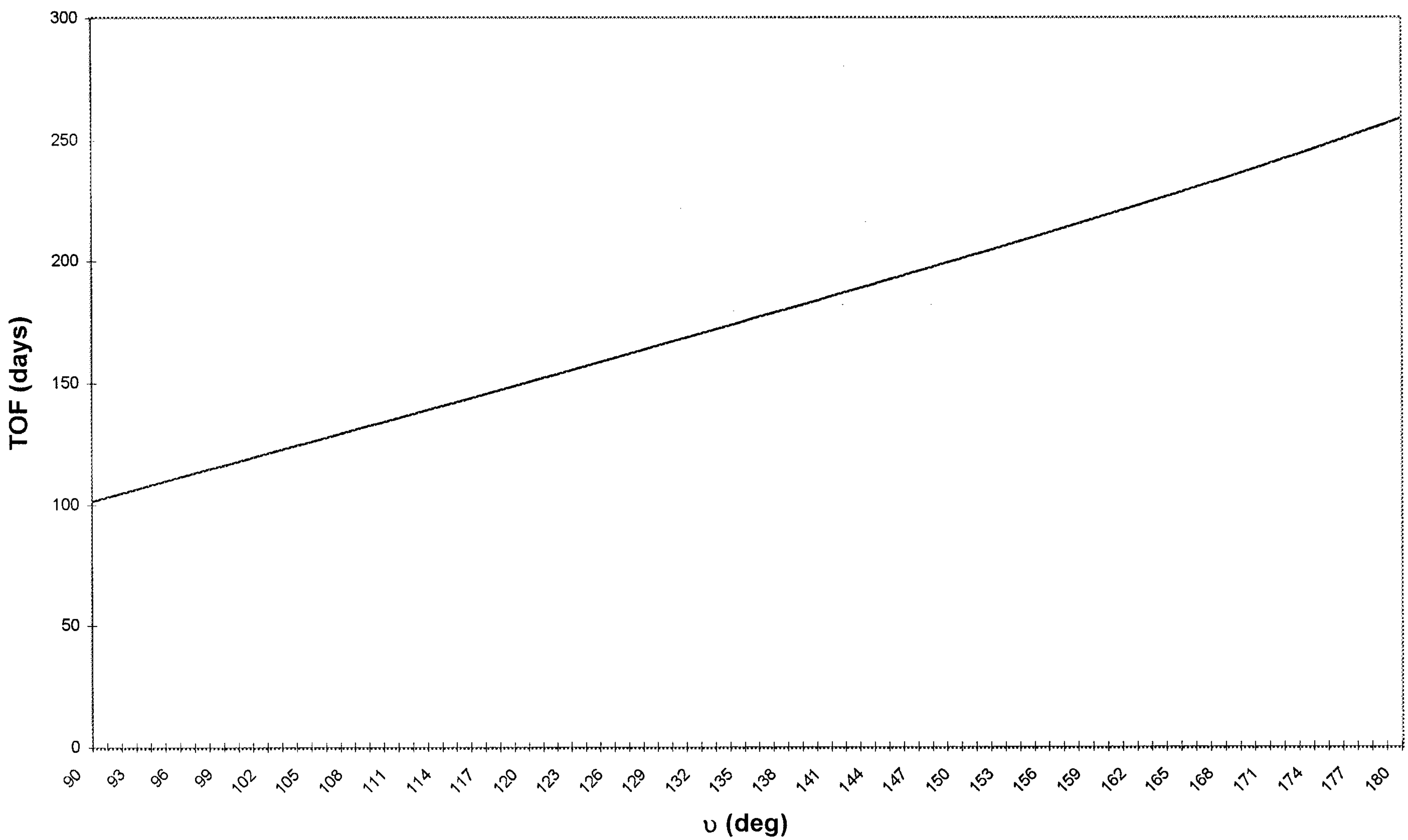
$$\boxed{\Delta V_{tot} = 6.35644\ Km/s}$$

R-1	0.6562	
R init	1	AU
R fin	1.524	AU
μ	1	
V init	1	AU/TU
V fin	0.810042	AU/TU

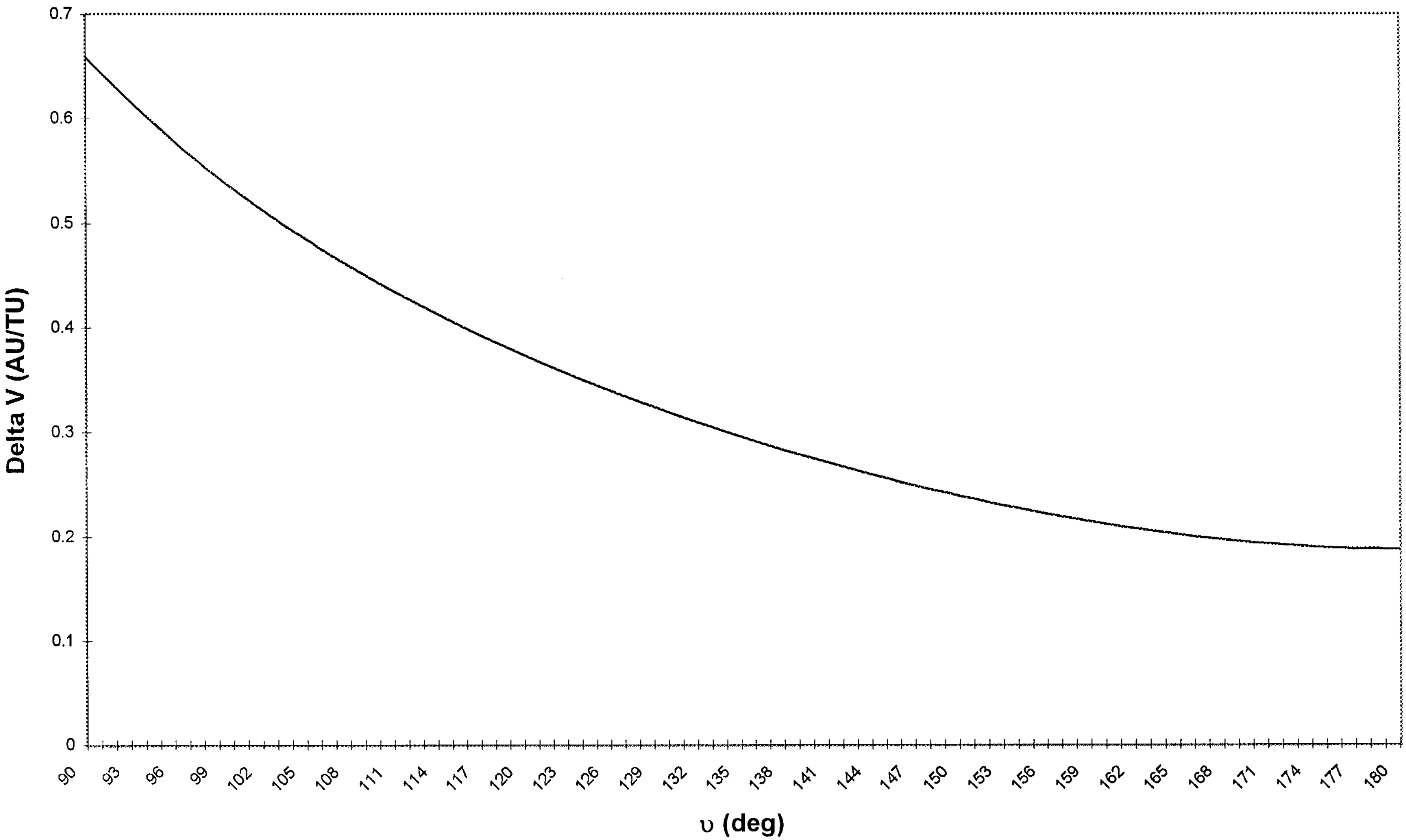
υ tx b	υ tx b	e trans	a trans	V tx a	V tx b	ΔV a	ϕ tx b	ΔV b	ΔV tot	E	TOF (days)
90	1.570796	0.523926	2.100512	1.234474	0.914473	0.234474	0.482604	0.424396	0.65887	1.019343	101.4205602
91	1.58825	0.510352	2.042284	1.228964	0.907022	0.228964	0.475465	0.415207	0.644171	1.050278	103.0750919
92	1.605703	0.497468	1.989924	1.223711	0.899891	0.223711	0.468396	0.406322	0.630033	1.080665	104.7251722
93	1.623156	0.485226	1.9426	1.218698	0.893063	0.218698	0.461395	0.397724	0.616423	1.110549	106.3711664
94	1.640609	0.473582	1.899631	1.213912	0.88652	0.213912	0.454461	0.389397	0.603309	1.13997	108.0134213
95	1.658063	0.462497	1.860456	1.209338	0.880246	0.209338	0.447593	0.381325	0.590662	1.168962	109.6522661
96	1.675516	0.451935	1.824602	1.204963	0.874226	0.204963	0.440789	0.373494	0.578457	1.197559	111.2880146
97	1.692969	0.441863	1.791675	1.200776	0.868446	0.200776	0.434049	0.365892	0.566668	1.225787	112.920966
98	1.710423	0.43225	1.761339	1.196766	0.862894	0.196766	0.42737	0.358507	0.555274	1.253671	114.5514066
99	1.727876	0.423068	1.733308	1.192924	0.857557	0.192924	0.420752	0.351327	0.544252	1.281236	116.1796103
100	1.745329	0.414293	1.707337	1.18924	0.852425	0.18924	0.414193	0.344343	0.533583	1.3085	117.8058403
101	1.762783	0.405899	1.683215	1.185706	0.847487	0.185706	0.407693	0.337544	0.52325	1.335484	119.4303492
102	1.780236	0.397865	1.660758	1.182314	0.842734	0.182314	0.401249	0.330922	0.513236	1.362203	121.0533808
103	1.797689	0.390171	1.639805	1.179055	0.838157	0.179055	0.394861	0.324469	0.503524	1.388674	122.6751701
104	1.815142	0.382799	1.620217	1.175925	0.833747	0.175925	0.388528	0.318176	0.494101	1.414911	124.2959444
105	1.832596	0.37573	1.601871	1.172915	0.829497	0.172915	0.382249	0.312037	0.484952	1.440927	125.9159237
106	1.850049	0.368949	1.584657	1.170021	0.8254	0.170021	0.376021	0.306044	0.476065	1.466734	127.5353216
107	1.867502	0.36244	1.568479	1.167236	0.821447	0.167236	0.369845	0.300192	0.467428	1.492343	129.1543457
108	1.884956	0.356189	1.553252	1.164555	0.817634	0.164555	0.363719	0.294474	0.459029	1.517765	130.7731981
109	1.902409	0.350185	1.538898	1.161974	0.813954	0.161974	0.357641	0.288884	0.450859	1.543009	132.3920758
110	1.919862	0.344413	1.525351	1.159488	0.8104	0.159488	0.351612	0.283419	0.442907	1.568084	134.0111714
111	1.937315	0.338863	1.512547	1.157093	0.806969	0.157093	0.34563	0.278071	0.435164	1.592999	135.6306735
112	1.954769	0.333525	1.500432	1.154784	0.803655	0.154784	0.339694	0.272838	0.427622	1.617761	137.2507666
113	1.972222	0.328388	1.488956	1.152557	0.800453	0.152557	0.333802	0.267714	0.420272	1.642377	138.8716323
114	1.989675	0.323444	1.478073	1.15041	0.797358	0.15041	0.327955	0.262696	0.413106	1.666855	140.4934488
115	2.007129	0.318682	1.467743	1.148339	0.794366	0.148339	0.32215	0.257779	0.406118	1.6912	142.1163918
116	2.024582	0.314096	1.457929	1.14634	0.791474	0.14634	0.316388	0.25296	0.3993	1.715419	143.7406348
117	2.042035	0.309677	1.448596	1.144411	0.788678	0.144411	0.310666	0.248235	0.392647	1.739518	145.3663489
118	2.059489	0.305418	1.439714	1.142549	0.785973	0.142549	0.304985	0.243602	0.386151	1.763501	146.9937037
119	2.076942	0.301312	1.431254	1.140751	0.783357	0.140751	0.299342	0.239057	0.379808	1.787374	148.6228671
120	2.094395	0.297353	1.423191	1.139014	0.780826	0.139014	0.293738	0.234597	0.373612	1.811141	150.2540059
121	2.111848	0.293536	1.415499	1.137337	0.778377	0.137337	0.288172	0.23022	0.367557	1.834807	151.8872856
122	2.129302	0.289853	1.408159	1.135717	0.776008	0.135717	0.282642	0.225923	0.36164	1.858376	153.5228711
123	2.146755	0.2863	1.401149	1.134152	0.773716	0.134152	0.277147	0.221704	0.355856	1.881852	155.160927
124	2.164208	0.282871	1.39445	1.132639	0.771497	0.132639	0.271688	0.21756	0.3502	1.905239	156.801617
125	2.181662	0.279563	1.388046	1.131178	0.76935	0.131178	0.266262	0.21349	0.344668	1.928541	158.4451052
126	2.199115	0.27637	1.381921	1.129765	0.767272	0.129765	0.26087	0.209491	0.339257	1.95176	160.0915554
127	2.216568	0.273288	1.37606	1.1284	0.76526	0.1284	0.25551	0.205562	0.333962	1.974901	161.7411321
128	2.234021	0.270312	1.37045	1.127081	0.763314	0.127081	0.250182	0.2017	0.328782	1.997966	163.3939998
129	2.251475	0.26744	1.365077	1.125807	0.76143	0.125807	0.244884	0.197905	0.323712	2.020959	165.0503241
130	2.268928	0.264668	1.359929	1.124574	0.759608	0.124574	0.239617	0.194175	0.318749	2.043882	166.7102713
131	2.286381	0.261991	1.354997	1.123384	0.757844	0.123384	0.234378	0.190508	0.313892	2.066737	168.3740087
132	2.303835	0.259407	1.350269	1.122233	0.756137	0.122233	0.229169	0.186903	0.309136	2.089529	170.0417051
133	2.321288	0.256913	1.345737	1.121121	0.754486	0.121121	0.223987	0.18336	0.304481	2.112258	171.7135304
134	2.338741	0.254505	1.34139	1.120047	0.752888	0.120047	0.218832	0.179876	0.299923	2.134928	173.3896564
135	2.356194	0.252181	1.337222	1.119009	0.751343	0.119009	0.213703	0.176451	0.29546	2.15754	175.0702566
136	2.373648	0.249938	1.333224	1.118006	0.749849	0.118006	0.2086	0.173085	0.291091	2.180098	176.7555067
137	2.391101	0.247774	1.329388	1.117038	0.748405	0.117038	0.203522	0.169776	0.286814	2.202602	178.4455842
138	2.408554	0.245686	1.325709	1.116103	0.747009	0.116103	0.198469	0.166524	0.282627	2.225056	180.1406693
139	2.426008	0.243673	1.322179	1.115201	0.74566	0.115201	0.193438	0.163328	0.278529	2.24746	181.8409446
140	2.443461	0.241731	1.318792	1.11433	0.744356	0.11433	0.188431	0.160188	0.274518	2.269818	183.5465956

141	2.460914	0.239858	1.315544	1.113489	0.743098	0.113489	0.183446	0.157104	0.270593	2.29213	185.2578108
142	2.478368	0.238054	1.312429	1.112679	0.741883	0.112679	0.178482	0.154075	0.266753	2.314399	186.9747815
143	2.495821	0.236315	1.309441	1.111897	0.74071	0.111897	0.17354	0.151101	0.262998	2.336626	188.6977028
144	2.513274	0.234641	1.306576	1.111144	0.739579	0.111144	0.168617	0.148182	0.259326	2.358813	190.4267731
145	**2.530727**	**0.233029**	**1.30383**	**1.110418**	**0.738488**	**0.110418**	**0.163715**	**0.145319**	**0.255737**	**2.380962**	**192.1621946**
146	2.548181	0.231478	1.301199	1.10972	0.737438	0.10972	0.158831	0.142511	0.252231	2.403073	193.9041737
147	2.565634	0.229986	1.298678	1.109048	0.736425	0.109048	0.153966	0.139759	0.248807	2.425149	195.6529207
148	2.583087	0.228553	1.296265	1.108401	0.735451	0.108401	0.149118	0.137063	0.245464	2.447191	197.4086505
149	2.600541	0.227176	1.293955	1.10778	0.734514	0.10778	0.144288	0.134424	0.242204	2.4692	199.1715827
150	2.617994	0.225854	1.291745	1.107183	0.733614	0.107183	0.139474	0.131843	0.239026	2.491178	200.9419418
151	2.635447	0.224586	1.289633	1.10661	0.732749	0.10661	0.134676	0.12932	0.23593	2.513126	202.7199575
152	2.6529	0.22337	1.287615	1.106061	0.73192	0.106061	0.129894	0.126856	0.232917	2.535045	204.5058649
153	2.670354	0.222207	1.285689	1.105535	0.731124	0.105535	0.125126	0.124452	0.229987	2.556937	206.2999046
154	2.687807	0.221094	1.283852	1.105031	0.730363	0.105031	0.120373	0.12211	0.227141	2.578802	208.1023235
155	2.70526	0.220031	1.282102	1.10455	0.729635	0.10455	0.115633	0.11983	0.22438	2.600643	209.9133745
156	2.722714	0.219016	1.280437	1.104091	0.728939	0.104091	0.110907	0.117615	0.221705	2.62246	211.7333172
157	2.740167	0.21805	1.278854	1.103653	0.728276	0.103653	0.106193	0.115465	0.219118	2.644254	213.562418
158	2.75762	0.21713	1.277351	1.103236	0.727644	0.103236	0.101491	0.113382	0.216618	2.666026	215.4009505
159	2.775074	0.216256	1.275927	1.10284	0.727043	0.10284	0.0968	0.111369	0.214209	2.687778	217.2491958
160	2.792527	0.215428	1.27458	1.102465	0.726474	0.102465	0.092121	0.109426	0.211891	2.709511	219.1074428
161	2.80998	0.214644	1.273309	1.102109	0.725934	0.102109	0.087451	0.107557	0.209666	2.731225	220.9759887
162	2.827433	0.213905	1.272111	1.101774	0.725425	0.101774	0.082792	0.105762	0.207536	2.752922	222.8551394
163	2.844887	0.213209	1.270985	1.101458	0.724945	0.101458	0.078142	0.104046	0.205503	2.774603	224.7452096
164	2.86234	0.212555	1.26993	1.101161	0.724494	0.101161	0.0735	0.102409	0.203569	2.796268	226.6465236
165	2.879793	0.211944	1.268946	1.100883	0.724072	0.100883	0.068867	0.100854	0.201737	2.817919	228.5594155
166	2.897247	0.211375	1.268029	1.100625	0.723679	0.100625	0.064241	0.099383	0.200008	2.839557	230.4842295
167	2.9147	0.210847	1.267181	1.100385	0.723314	0.100385	0.059623	0.098	0.198385	2.861182	232.421321
168	2.932153	0.210359	1.266399	1.100163	0.722977	0.100163	0.055011	0.096707	0.19687	2.882796	234.3710563
169	2.949606	0.209912	1.265682	1.09996	0.722667	0.09996	0.050406	0.095506	0.195466	2.904399	236.3338137
170	2.96706	0.209505	1.265031	1.099775	0.722386	0.099775	0.045805	0.094399	0.194174	2.925992	238.3099838
171	2.984513	0.209138	1.264444	1.099608	0.722132	0.099608	0.04121	0.093389	0.192997	2.947577	240.2999702
172	3.001966	0.208811	1.26392	1.099459	0.721905	0.099459	0.03662	0.092478	0.191938	2.969154	242.3041898
173	3.01942	0.208522	1.263459	1.099328	0.721705	0.099328	0.032034	0.091669	0.190997	2.990724	244.323074
174	3.036873	0.208273	1.263061	1.099214	0.721532	0.099214	0.027451	0.090964	0.190178	3.012288	246.3570689
175	3.054326	0.208062	1.262725	1.099119	0.721386	0.099119	0.022871	0.090363	0.189482	3.033847	248.406636
176	3.071779	0.207889	1.26245	1.09904	0.721266	0.09904	0.018294	0.08987	0.18891	3.055401	250.4722535
177	3.089233	0.207756	1.262237	1.098979	0.721174	0.098979	0.013719	0.089485	0.188464	3.076952	252.5544161
178	3.106686	0.20766	1.262084	1.098936	0.721107	0.098936	0.009145	0.089209	0.188145	3.0985	254.6536371
179	3.124139	0.207603	1.261993	1.09891	0.721068	0.09891	0.004572	0.089043	0.187953	3.120047	256.7704479
180	3.141593	0.207584	1.261963	1.098901	0.721054	0.098901	3.21E-17	0.088987	0.187889	3.141593	258.9054002

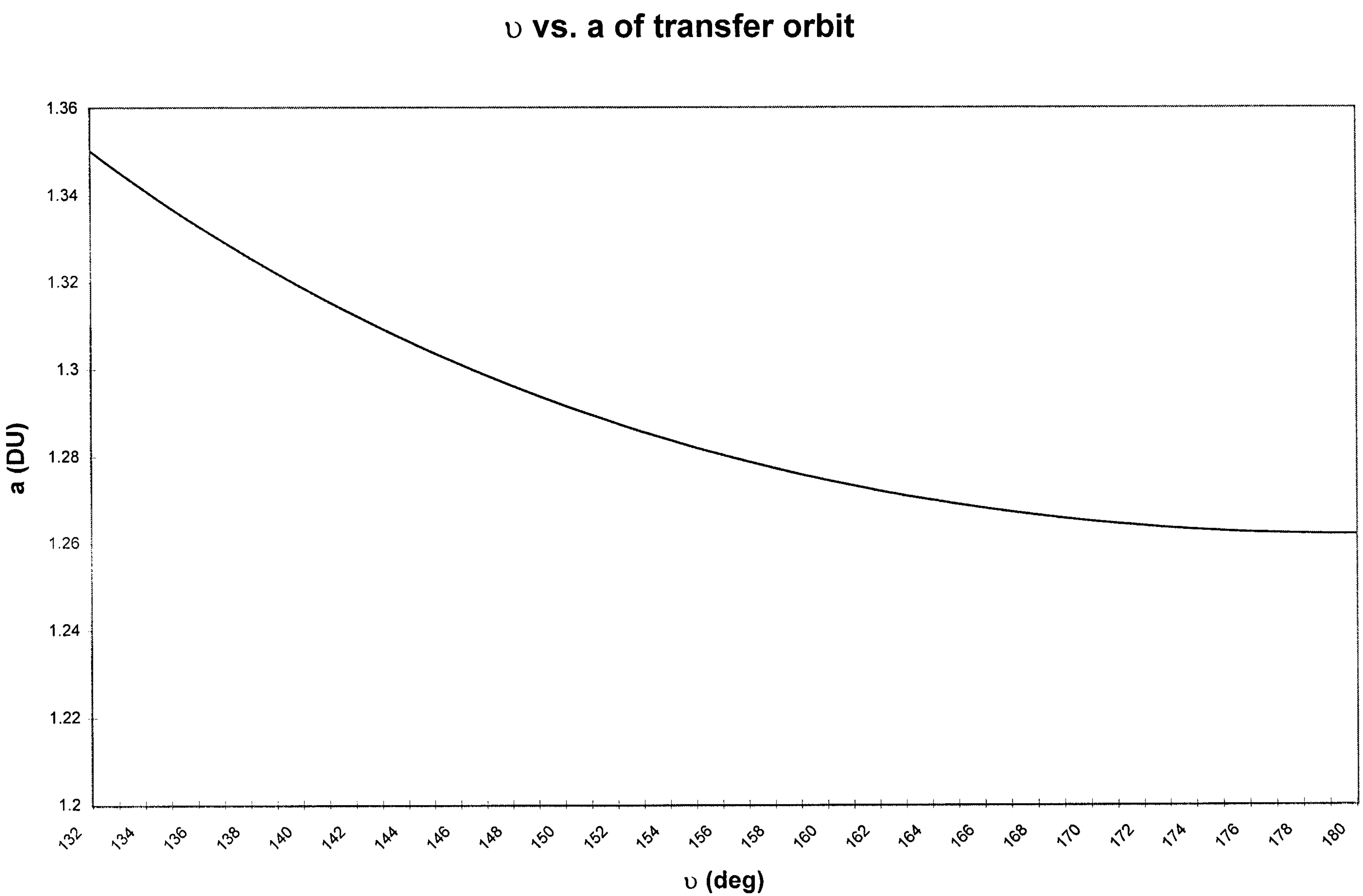
Time of Flight vs. υ

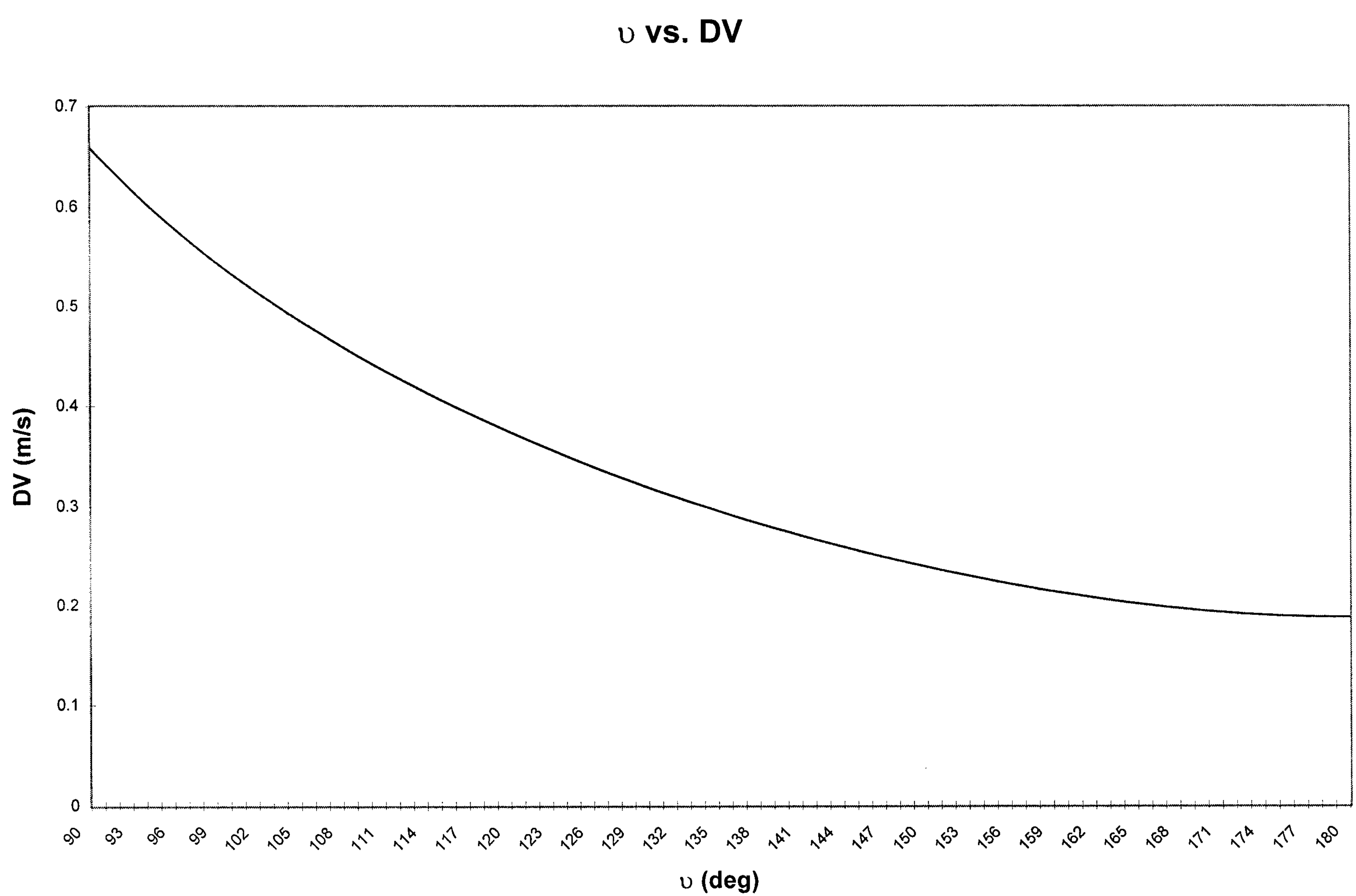


Delta V vs. υ



Delta V (AU/TU)
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
90
93
96
99
102
105
108
111
114
117
120
123
126
129
132
135
138
141
144
147
150
153
156
159
162
165
168
171
174
177
180
υ (deg)


υ vs. a of transfer orbit
a (DU)
1.36
1.34
1.32
1.3
1.28
1.26
1.24
1.22
1.2
132
134
136
138
140
142
144
146
148
150
152
154
156
158
160
162
164
166
168
170
172
174
176
178
180
υ (deg)



υ vs. DV
DV (m/s)
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
90
93
96
99
102
105
108
111
114
117
120
123
126
129
132
135
138
141
144
147
150
153
156
159
162
165
168
171
174
177
180
υ (deg)

Appendix B: Preliminary Mission Analysis using Dummkopf Charts

Information to determine the Dummkopf chart for Earth Escape

M payload	94046	kg
dV	3290	m/s
gravity	9.81	m/s^2

f inert 0.025

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	157628.5	4041.756	255716.3	98087.76	90.91435	Yes
360	150636.1	3862.464	248544.5	97908.46		Yes
370	144213.3	3697.778	241957.1	97743.78		Yes
380	138295.2	3546.031	235887.2	97592.03		Yes
390	132826	3405.794	230277.8	97451.79		Yes
400	127757.6	3275.837	225079.5	97321.84		Yes
410	123048 8	3155 097	220249 9	97201.1		Yes
420	118663.3	3042.649	215752	97088 65		Yes
430	114569 8	2937 687	211553.5	96983 69		Yes
440	110740.7	2839.504	207626 2	96885 5		Yes
450	107151.6	2747 478	203945.1	96793.48		Yes
460	103781.3	2661.059	200488 3	96707 06		Yes
470	100610 6	2579 76	197236 4	96625.76		Yes
480	97622 78	2503.148	194171.9	96549.15		Yes
490	94802 67	2430.838	191279.5	96476.84		Yes
500	92136.82	2362.483	188545.3	96408.48		Yes
510	89613.15	2297.773	185956.9	96343.77		Yes
520	87220.78	2236.43	183503.2	96282.43		Yes

f inert 0.05

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	165097.2	8689.327	267832.5	102735.3	111.9499	Yes
360	157442.6	8286.451	259775	102332.5		Yes
370	150439.8	7917.883	252403.7	101963.9		Yes
380	144011	7579.525	245636.5	101625.5		Yes
390	138090	7267.893	239403.9	101313.9		Yes
400	132620.2	6980.012	233646.2	101026		Yes
410	127553.2	6713.326	228312 5	100759.3		Yes
420	122846.9	6465.628	223358.6	100511.6		Yes
430	118465	6235.001	218746	100281		Yes
440	114375.7	6019 775	214441.5	100065.8		Yes
450	110551.3	5818.488	210415.8	99864.49		Yes
460	106967.3	5629 855	206643.1	99675.86		Yes
470	103602.1	5452.742	203100.8	99498.74		Yes
480	100436.7	5286.143	199768.9	99332.14		Yes
490	97454.18	5129.167	196629.3	99175.17		Yes
500	94639.34	4981.018	193666.4	99027.02		Yes
510	91978.69	4840.984	190865.7	98886.98		Yes
520	89460.12	4708.427	188214.5	98754.43		Yes

f inert 0.075

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	173776.1	14089.95	281912.1	108136	129.4739	Yes
360	165316.1	13404.01	272766.1	107450		Yes
370	157612.6	12779.4	264438	106825.4		Yes
380	150570.5	12208.42	256824.9	106254.4		Yes
390	144109.9	11684.59	249840.5	105730.6		Yes
400	138163.1	11202.42	243411.5	105248.4		Yes
410	132672.5	10757.23	237475.7	104803.2		Yes
420	127588.4	10345	231979.4	104391		Yes
430	122868.2	9962 284	226876.4	104008.3		Yes
440	118474.9	9606.071	222127	103652.1		Yes
450	114376.3	9273.754	217696.1	103319.8		Yes
460	110544.3	8963.049	213553.3	103009		Yes
470	106954.1	8671.951	209672	102718		Yes
480	103583.9	8398.693	206028.6	102444.7		Yes
490	100414.4	8141.712	202602.2	102187.7		Yes
500	97428.62	7899.618	199374.2	101945.6		Yes
510	94611.17	7671.176	196328.3	101717.2		Yes
520	91948.45	7455.28	193449.7	101501.3		Yes

f inert 0.1

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	183985.2	20442.8	298474	114488.8	145.6502	Yes
360	174529.1	19392.12	287967.2	113438.1		Yes
370	165965.2	18440.58	278451.8	112486.6		Yes
380	158175.4	17575.04	269796.4	111621		Yes
390	151061.1	16784.57	261891.7	110830.6		Yes
400	144539.8	16059.98	254645.8	110106		Yes
410	138541.6	15393.51	247981.1	109439.5		Yes
420	133007.1	14778.57	241831.7	108824.6		Yes
430	127885.5	14209.5	236141	108255.5		Yes
440	123133.1	13681.45	230860.5	107727.5		Yes
450	118711.9	13190.21	225948.1	107236.2		Yes
460	114589.1	12732.12	221367.2	106778.1		Yes
470	110735.9	12303.99	217085.9	106350		Yes
480	107127.2	11903.02	213076.2	105949		Yes
490	103740.7	11526.75	209313.5	105572.7		Yes
500	100557	11173	205776	105219		Yes
510	97558.45	10839.83	202444.3	104885.8		Yes
520	94729.74	10525.53	199301.3	104571.5		Yes

f inert 0.125

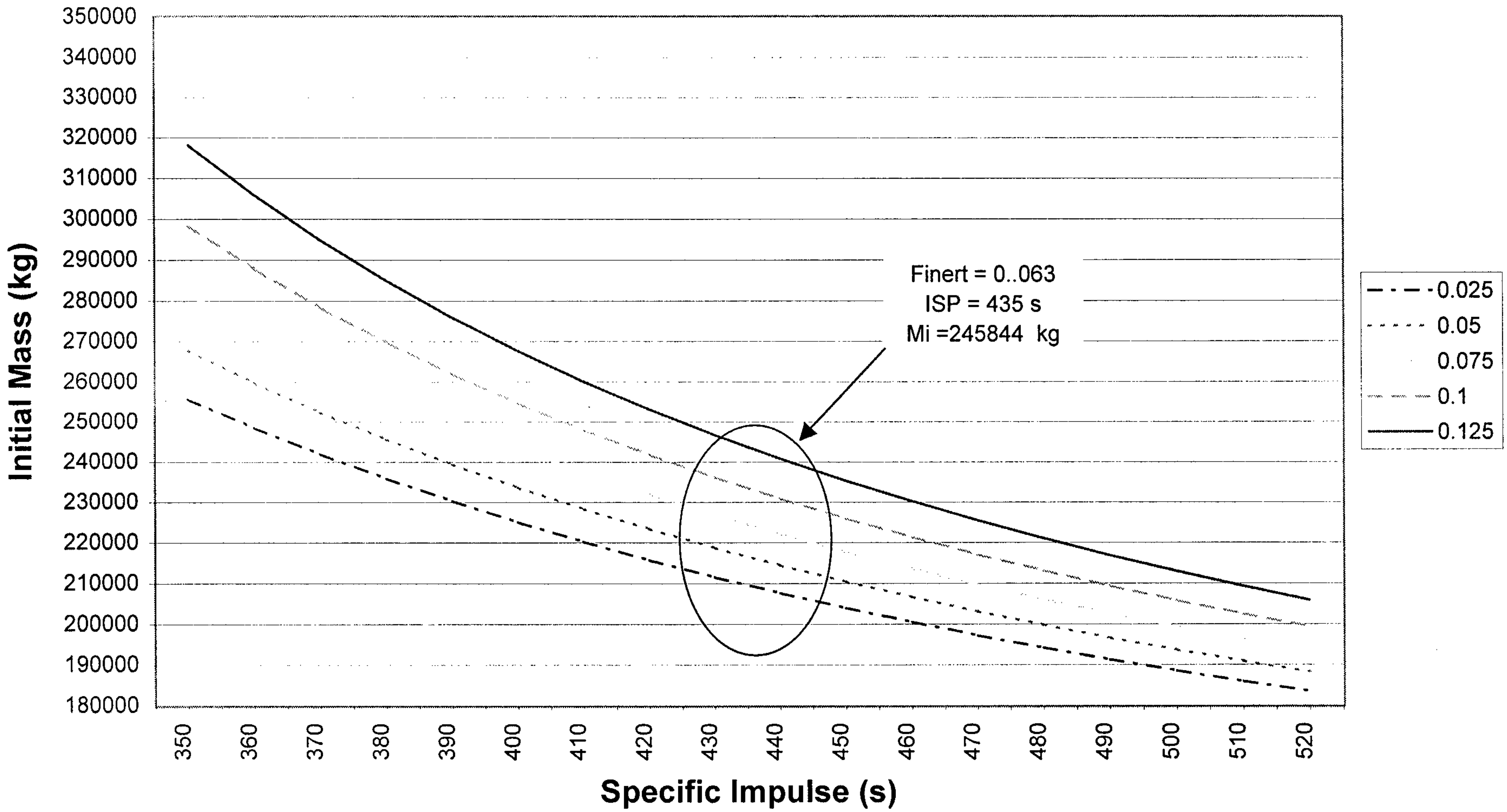
ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	196168.4	28024.06	318238.5	122070.1	161.2799	Yes
360	185454.9	26493.56	305994.5	120539.6		Yes
370	175814.9	25116.41	294977.3	119162.4		Yes
380	167097.3	23871.04	285014.3	117917		Yes
390	159177.9	22739.7	275963.6	116785.7		Yes
400	151953.7	21707.68	267707.4	115753.7		Yes
410	145338.5	20762.64	260147.1	114808.6		Yes
420	139259.6	19894.22	253199.8	113940.2		Yes
430	133655.3	19093.61	246794.9	113139.6		Yes
440	128473	18353.28	240872.3	112399.3		Yes
450	123667.5	17666.79	235380.3	111712.8		Yes
460	119199.8	17028.54	230274.3	111074.5		Yes
470	115035.9	16433.7	225515.6	110479.7		Yes
480	111146.4	15878.06	221070.5	109924.1		Yes
490	107505.4	15357.92	216909.3	109403.9		Yes
500	104090.2	14870.03	213006.2	108916		Yes
510	100880.6	14411.52	209338.1	108457.5		Yes
520	97858.96	13979.85	205884.8	108025.9		Yes

f inert 0.063

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
435	142233.4	9563.183	245842.6	103609.2		Yes

Dummkopf Chart (Earth Escape)

Initial Mass vs. ISP, Payload = 94046 kg, Delta V = 3290 m/s



Information to determine the Dummkopf chart for Mars Insertion

M payload	41000	kg
dV	3067	m/s
gravity	9.81	m/s^2

f inert 0.05

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	64029.19	3369.957	108399.1	44369.96	104.3619	Yes
360	61164.02	3219.159	105383.2	44219.16		Yes
370	58535.63	3080.822	102616.4	44080.82		Yes
380	56116.46	2953.498	100070	43953.5		Yes
390	53883.02	2835.948	97718.97	43835.95		Yes
400	51815.16	2727.113	95542.27	43727.11		Yes
410	49895.49	2626.078	93521.57	43626.08		Yes
420	48108.95	2532.05	91641	43532.05		Yes
430	46442.42	2444.338	89886.76	43444.34		Yes
440	44884.43	2362.339	88246.77	43362.34		Yes
450	43424.9	2285.521	86710.42	43285.52		Yes
460	42054.95	2213.419	85268.37	43213.42		Yes
470	40766.72	2145.617	83912.34	43145.62		Yes
480	39553.23	2081.749	82634.98	43081.75		Yes
490	38408.28	2021.488	81429.76	43021.49		Yes
500	37326.29	1964.542	80290.84	42964.54		Yes
510	36302.31	1910.648	79212.96	42910.65		Yes
520	35331.86	1859.572	78191.43	42859.57		Yes
530	34410.91	1811.101	77222.01	42811.1		Yes
540	33535.82	1765.043	76300.87	42765.04		Yes
550	32703.3	1721.227	75424.53	42721.23		Yes
560	31910.37	1679.493	74589.86	42679.49		Yes

f inert 0.1

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	70464.48	7829.387	119293.9	48829.39	135.7779	Yes
360	67009.98	7445.553	115455.5	48445.55		Yes
370	63868.04	7096.449	111964.5	48096.45		Yes
380	60998.84	6777.649	108776.5	47777.65		Yes
390	58368.96	6485.44	105854.4	47485.44		Yes
400	55950.18	6216.687	103166.9	47216.69		Yes
410	53718.5	5968.722	100687.2	46968.72		Yes
420	51653.37	5739.263	98392.63	46739.26		Yes
430	49737.12	5526.347	96263.47	46526.35		Yes
440	47954.48	5328.276	94282.76	46328.28		Yes
450	46292.16	5143.573	92435.73	46143.57		Yes
460	44738.56	4970.951	90709.51	45970.95		Yes
470	43283.52	4809.28	89092.8	45809.28		Yes
480	41918.08	4657.565	87575.65	45657.56		Yes
490	40634.34	4514.927	86149.27	45514.93		Yes
500	39425.28	4380.587	84805.87	45380.59		Yes
510	38284.66	4253.851	83538.51	45253.85		Yes
520	37206.9	4134.1	82341	45134.1		Yes
530	36187.02	4020.78	81207.8	45020.78		Yes
540	35220.53	3913.392	80133.92	44913.39		Yes
550	34303.41	3811.49	79114.9	44811.49		Yes
560	33432.01	3714.668	78146.68	44714.67		Yes

f inert 0.15

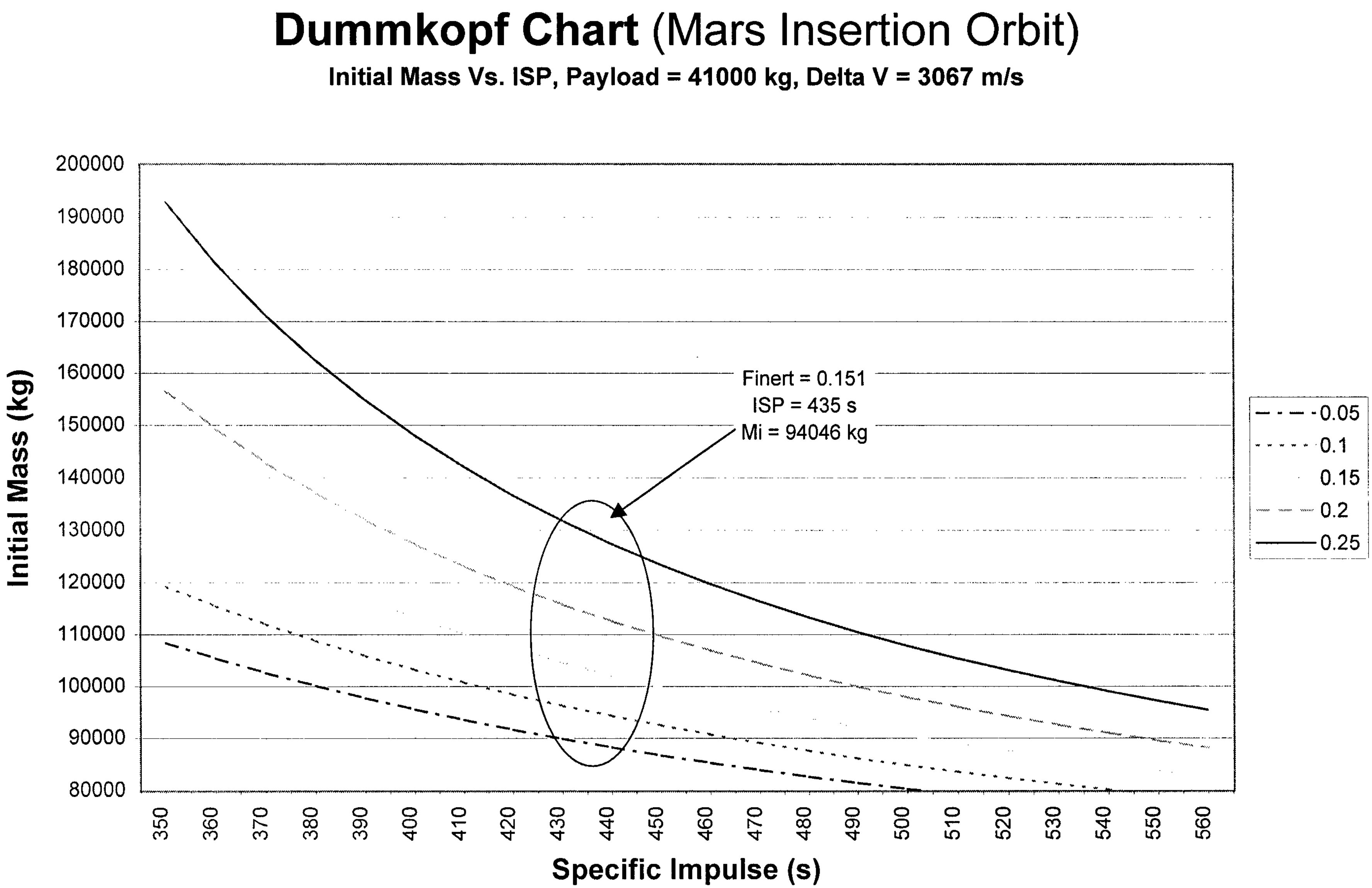
ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	79381.38	14008.48	134389.9	55008.48	164.7973	Yes
360	75024.29	13239.58	129263.9	54239.58		Yes
370	71107.83	12548.44	124656.3	53548.44		Yes
380	67569.3	11923.99	120493.3	52923.99		Yes
390	64357.27	11357.17	116714.4	52357.17		Yes
400	61429.18	10840.44	113269.6	51840.44		Yes
410	58749.48	10367.55	110117	51367.55		Yes
420	56288.28	9933.225	107221.5	50933.23		Yes
430	54020.26	9532.987	104553.2	50532.99		Yes
440	51923.84	9163.031	102086.9	50163.03		Yes
450	49980.51	8820.091	99800.61	49820.09		Yes
460	48174.32	8501.35	97675.67	49501.35		Yes
470	46491.41	8204.367	95695.78	49204.37		Yes
480	44919.76	7927.016	93846.77	48927.02		Yes
490	43448.81	7667.437	92116.24	48667.44		Yes
500	42069.3	7423.994	90493.3	48423.99		Yes
510	40773.07	7195.248	88968.32	48195.25		Yes
520	39552.89	6979.922	87532.81	47979.92		Yes
530	38402.33	6776.882	86179.21	47776.88		Yes
540	37315.67	6585.117	84900.78	47585.12		Yes
550	36287.78	6403.726	83691.51	47403.73		Yes
560	35314.08	6231.897	82545.98	47231.9		Yes

f inert 0.2

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	92558.2	23139.55	156697.7	64139.55	194.2543	Yes
360	86688.04	21672.01	149360	62672.01		Yes
370	81501.26	20375.31	142876.6	61375.31		Yes
380	76886.28	19221.57	137107.9	60221.57		Yes
390	72754.47	18188.62	131943.1	59188.62		Yes
400	69034.51	17258.63	127293.1	58258.63		Yes
410	65668.39	16417.1	123085.5	57417.1		Yes
420	62608.44	15652.11	119260.5	56652.11		Yes
430	59815.15	14953.79	115768.9	55953.79		Yes
440	57255.49	14313.87	112569.4	55313.87		Yes
450	54901.63	13725.41	109627	54725.41		Yes
460	52729.97	13182.49	106912.5	54182.49		Yes
470	50720.36	12680.09	104400.5	53680.09		Yes
480	48855.52	12213.88	102069.4	53213.88		Yes
490	47120.49	11780.12	99900.61	52780.12		Yes
500	45502.32	11375.58	97877.9	52375.58		Yes
510	43989.71	10997.43	95987.13	51997.43		Yes
520	42572.75	10643.19	94215.94	51643.19		Yes
530	41242.75	10310.69	92553.44	51310.69		Yes
540	39992.01	9998.002	90990.01	50998		Yes
550	38813.72	9703.43	89517.15	50703.43		Yes
560	37701.82	9425.456	88127.28	50425.46		Yes

f inert 0.25

ISP	Mprop	Minert	Mi	Mf	Min ISP	Feasible
350	114005.7	38001.89	193007.6	79001.89	225.5222	Yes
360	105228.9	35076.29	181305.1	76076.29		Yes
370	97682.68	32560.89	171243.6	73560.89		Yes
380	91126.96	30375.65	162502.6	71375.65		Yes
390	85380.04	28460.01	154840.1	69460.01		Yes
400	80302.01	26767.34	148069.3	67767.34		Yes
410	75783.37	25261.12	142044.5	66261.12		Yes
420	71737.21	23912.4	136649.6	64912.4		Yes
430	68093.67	22697.89	131791.6	63697.89		Yes
440	64795.99	21598.66	127394.6	62598.66		Yes
450	61797.53	20599.18	123396.7	61599.18		Yes
460	59059.67	19686.56	119746.2	60686.56		Yes
470	56550.12	18850.04	116400.2	59850.04		Yes
480	54241.71	18080.57	113322.3	59080.57		Yes
490	52111.37	17370.46	110481.8	58370.46		Yes
500	50139.43	16713.14	107852.6	57713.14		Yes
510	48309.01	16103	105412	57103		Yes
520	46605.53	15535.18	103140.7	56535.18		Yes
530	45016.32	15005.44	101021.8	56005.44		Yes
540	43530.35	14510.12	99040.47	55510.12		Yes
550	42137.97	14045.99	97183.96	55045.99		Yes
560	40830.67	13610.22	95440.89	54610.22		Yes


Dummkopf Chart (Mars Insertion Orbit)
Initial Mass Vs. ISP, Payload = 41000 kg, Delta V = 3067 m/s
Initial Mass (kg)
200000
190000
180000
170000
160000
150000
140000
130000
120000
110000
100000
90000
80000
350
360
370
380
390
400
410
420
430
440
450
460
470
480
490
500
510
520
530
540
550
560
Specific Impulse (s)
Finert = 0.151
ISP = 435 s
Mi = 94046 kg
0.05
0.1
0.15
0.2
0.25

Appendix C: Top Level Mission Analysis

IMPRESS
T1000G
Cryogenic Storage System

Lee Gentile
Mars mission design

Given

dV	3290	m/s	
Mass of Habitat Unit	30000	kg	
Mass of Cryo Unit	11000	kg	
Mass of Payload	94046.75849	kg	
Max G (F/W)	6	earth G's	
Thrust to Weight Ratio	1.2		
g0	9.81	m/s^2	

Thrust Requirement-top level analysis

ISP	435	sec	
finert	0.063		
mass prop	142234.5166	kg	add 20% for residual
max inert mass	9563.25992	kg	
max initial mass	245844.535	kg	
F	2894081.866	N	

Part 1: system mass and envelope

Me	3592.027051	kg
Le	15659.96208	cm
De	10346.35226	cm

Part 2: propellant
Oxygen and Hydrogen

O/F	3.8	
ISP	435	sec

Part 3 & 4:Engine cycle and cooling approach
Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000	Pa	selected	
Chamber pressure (Pc)	11500000	Pa	selected	
Atmospheric pressure(Pa)	650	Pa	vacuum	
γ	1.23		O/F = 3.8	
Tc	2800	K	O/F = 3.8	
MW	9.8	kg/kmol	O/F = 3.8	
c*	2402.69663	m/s	1.02	c* eff.

Output

Me	4.627484385	eq 3.100
ε	52.52696338	to get Pe
mdot propellant	678.1918207	eq 1.7
At	0.141694713	eq 3.133
Ae	7.442793014	using ε
mdot ox	536.901858	O/F = 3.8
mdot fuel	141.2899626	O/F = 3.8

Determination of Propellant Needed at Mars

Mpayload	41000	kg
Minert	9380.228	kg
Isp	435	sec
g0	9.81	m/s^2
Delta V	3067	m/s
f inert	0.151	selected
M prop	53046.76	kg
f inert	0.150259	calculated

c*	2402.697
gamma	1.23
g0	9.81
Pe	15000
Pa	650
lambda	0.95

Pc	Me	ε	ISP
0	#DIV/0!	#DIV/0!	#DIV/0!
500000	2.83838	4.991658	378.6091
1000000	3.220975	8.245424	394.7603
1500000	3.446272	11.12957	403.0735
2000000	3.607461	13.80014	408.5232
2500000	3.733469	16.32383	412.5138
3000000	3.837156	18.73714	415.6299
3500000	3.925385	21.06289	418.168
4000000	4.002261	23.31658	420.2979
4500000	4.070435	25.50935	422.1254
5000000	4.131721	27.64958	423.7208
5500000	4.187416	29.74377	425.1327
6000000	4.238481	31.79709	426.3964
6500000	4.285645	33.81375	427.538
7000000	4.329479	35.79724	428.5775
7500000	4.370433	37.75048	429.5304
8000000	4.408874	39.67596	430.4091
8500000	4.445101	41.57582	431.2235
9000000	4.479361	43.45192	431.9816
9500000	4.511865	45.30589	432.6903
10000000	4.542788	47.13918	433.355
10500000	4.572281	48.95304	433.9806
11000000	4.600476	50.74863	434.5711
11500000	**4.627484**	**52.52696**	**435.1298**
12000000	4.653406	54.28897	435.6599
12500000	4.678327	56.03549	436.1638
13000000	4.702325	57.76728	436.6439
13500000	4.725467	59.48504	437.102
14000000	4.747815	61.18941	437.5401
14500000	4.769423	62.88098	437.9596
15000000	4.79034	64.5603	438.362
15500000	4.81061	66.22786	438.7484
16000000	4.830274	67.88413	439.12
16500000	4.849367	69.52955	439.4778
17000000	4.867924	71.16452	439.8227
17500000	4.885973	72.78943	440.1555
18000000	4.903545	74.40461	440.477
18500000	4.920663	76.01041	440.7879
19000000	4.937351	77.60713	441.0888
19500000	4.953632	79.19508	441.3802
20000000	4.969526	80.77452	441.6628
20500000	4.98505	82.34573	441.9369
21000000	5.000222	83.90893	442.2031
21500000	5.015059	85.46438	442.4617
22000000	5.029576	87.01229	442.7131
22500000	5.043786	88.55287	442.9577
23000000	5.057703	90.08632	443.1958
23500000	5.071339	91.61283	443.4278
24000000	5.084705	93.13259	443.6538
24500000	5.097813	94.64576	443.8742
25000000	5.110672	96.15251	444.0893

Estimate Masses and size tanks

Isp	435	sec
f inert	0.063	
payload	94046.76	kg
dV	3290	m/s

Estimate Masses

20% added for blow down extra

Mprop	142234.5	eq. 1.27	
Minert	9563.26	eq. 1.24	
Mfinal	103610		
Minitial	245844.5		
Mfuel	29632.19	O/F = 3.8	
Mox	112602.3	O/F = 3.8	
Vfuel	417.3548		71
Vox	98.60099		1142

Size Tanks

	spherical tanks Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	112602.3256	29632.19096
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.16E+09	1.16E+09
Tank Density (kg/m^3)	1550	1550
Vol. (5% Ullage)	103.531035	438.2225423
radius (m)	2.91291167	4.711964772
area (m^2)	106.6263382	279.0062512
wall thickness (m)	0.005022261	0.008124077
mass tank (kg)	830.0332974	3513.335899
tank factor	50800	50800
mass tank using TF	830.9938834	3517.401834

Thrust Chamber

N2O4/RP-1

Input

Pc (Pa)	11500000	
c*	2402.69663	
gamma	1.23	
At	0.141694713	
Ae	7.442793014	
L* Table 5.6	0.5	
Mc	0.4	
Ftu	3.10E+08	columbium
mult. factor	3	
cham cont angle	45	
dens comb cham	8500	columbium
expansion ratio	52.52696338	

Output

Ac (m^2)	0.329392156
Chamber Len (m)	0.215085136
Chamber Dia (m)	0.647607225
Troat Dia (m)	0.424748528
Len:Dia ratio	0.332122817
Cham Thick (m)	0.036036209
Mass Cham (kg)	169.5331742

Nozzle

Input

Nozzle Throat Dia	0.424749	
Ae	7.442793	
Nozzle exit Diam.	3.078386	
Cone 1/2 angle (deg)	20	
Lf for eff=.985 fig 5.25	0.6	
thick. throat wall (m)	0.018018	
Using 1/2 chamber wall		
thick. nozzle exit	0.009009	
density nozzle mater.	8500	columbium

Output

Ln (m) 15 deg nozzle	3.645404
Ln bell nozzle (m)	2.187242
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	1843.324

Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	1266.132

Mass Summary

Total engine mass	2871.331097
(using 50% for chamber & nozzle)	
Mass injectors (15%)	717.8327743
mass oxidizer tank	830.9938834
mass fuel tank	3517.401834
support structure	721.9726815
(10% of tank & engine masses)	
feed system	720.695954
(system level est.-engine mass)	
Total Inert mass	9380.228224
propellant mass	142234.5166
payload mass	94046.75849
Final mass	103426.9867
Inert mass fraction	0.061868839
Initial mass of vehicle	245661.5033
Thrust	2894081.866
F/W	1.200894068
Check G limit	2.852383613

Appendix D: Top Level Analysis Using Different Technology Combinations

IMPRESS, T1000G without cryogenics

IMPRESS, Cryogenics with Titanium tanks

Cryogenics, T1000G tanks without IMPRESS

Cryogenics, Titanium tanks without IMPRESS

Lee Gentile
Mars mission design
IMPRESS, T1000G tanks without Cryogenic Storage

Given

dV	3290	m/s	
Mass of Habitat Unit	30000	kg	
Mass of Cryo Unit	11000	kg	
Mass of Payload	4590614.889	kg	
Max G (F/W)	6	earth G's	
Thrust to Weight Ratio	1.2		
g0	9.81	m/s^2	

Thrust Requirement-top level analysis

ISP	435	sec	
finert	0.347		
mass prop	16728974.15	kg	add 20% for residual
max inert mass	8889669.264	kg	
max initial mass	30209258.3	kg	
F	355623388.7	N	

Part 1: system mass and envelope

Me	268955.1642	kg
Le	11145.76349	cm
De	8570.45574	cm

Part 2: propellant
Oxygen and Hydrogen

O/F	3.8	
ISP	435	sec

Part 3 & 4:Engine cycle and cooling approach
Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000	Pa	selected
Chamber pressure (Pc)	11500000	Pa	selected
Atmospheric pressure(Pa)	650	Pa	vacuum
γ	1.23		O/F = 3.8
Tc	2800	K	O/F = 3.8
MW	9.8	kg/kmol	O/F = 3.8
c*	2402.69663	m/s	1.02 c* eff.

Output

Me	4.627484385	eq 3.100
ε	52.52696338	to get Pe
mdot propellant	83335.88497	eq 1.7
At	17.41137826	eq 3.133
Ae	914.5668284	using ε
mdot ox	65974.24227	O/F = 3.8
mdot fuel	17361.6427	O/F = 3.8

Determination of Propellant Needed at Mars

Mpayload	41000	kg
Minert	9784866	kg
Isp	435	sec
g0	9.81	m/s^2
Delta V	3067	m/s
f inert	0.485	selected

M prop	4549615	kg

f inert	0.68261	calculated

Estimate Masses and size tanks

Isp	435	sec
f inert	0.347	
payload	4590615	kg
dV	3290	m/s

Estimate Masses

20% added for blow down extra

Mprop	16728974	eq. 1.27
Minert	8889669	eq. 1.24
Mfinal	13480284	
Minitial	30209258	
Mfuel	3485203	O/F = 3.8
Mox	13243771	O/F = 3.8
Vfuel	719171.6	4.846135207
Vox	172169	76.92307692

Size Tanks

	spherical tanks Composite Ox	Composite Fuel
Density Prop.	76.92307692	4.846135207
Mass Prop/Press.	13243771.2	3485202.948
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.16E+09	1.16E+09
Tank Density (kg/m^3)	1550	1550
Vol. (5% Ullage)	180777.4769	755130.2097
radius (m)	35.07668724	56.49080262
area (m^2)	15461.33553	40101.93738
wall thickness (m)	0.060477047	0.097397936
mass tank (kg)	1449336.668	6054061.164
tank factor	50800	50800
mass tank using TF	1451013.965	6061067.444

Density	Oxygen	Hydrogen
Temp Init (K)	100	100
Press Init	2000000	2000000
Press Fin	50000	50000
Temp Fin (K)	2.5	2.5
R	260	4127
Density init	76.92307692	4.846135207
Density fin	1.923076923	0.12115338

Thrust Chamber
N2O4/RP-1

Input

Pc (Pa)	11500000	
c*	2402.69663	
gamma	1.23	
At	17.41137826	
Ae	914.5668284	
L* Table 5.6	0.5	
Mc	0.4	
Ftu	3.10E+08	columbium
mult. factor	3	
cham cont angle	45	
dens comb cham	8500	columbium
expansion ratio	52.52696338	

Output

Ac (m^2)	40.47554979
Chamber Len (m)	0.215085136
Chamber Dia (m)	7.178793115
Troat Dia (m)	4.708381392
Len:Dia ratio	0.029961183
Cham Thick (m)	0.399465101
Mass Cham (kg)	64818.86625

Nozzle

Input

Nozzle Throat Dia	4.708381	
Ae	914.5668	
Nozzle exit Diam.	34.12422	
Cone 1/2 angle (deg)	20	
Lf for eff=.985 fig 5.25	0.6	
thick. throat wall (m)	0.199733	
Using 1/2 chamber wall		
thick. nozzle exit	0.099866	
density nozzle mater.	8500	columbium

Output

Ln (m) 15 deg nozzle	40.40968
Ln bell nozzle (m)	24.24581
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	2510851

Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	1724640

Mass Summary

Total engine mass (using 50% for chamber & nozzle)	3578917.058
Mass injectors (15%)	894729.2645
mass oxidizer tank	1451013.965
mass fuel tank	6061067.444
support structure (10% of tank & engine masses)	1109099.847
feed system (system level est.-engine mass)	-3309961.894
Total Inert mass	9784865.685
propellant mass	16728974.15
payload mass	4590614.889
Final mass	14375480.57
Inert mass fraction	0.369047477
Initial mass of vehicle	31104454.72
Thrust	355623388.7
F/W	1.165463606
Check G limit	2.521732041

Lee Gentile
Mars mission design
IMPRESS, Cryogenic Storage with Titanium Tanks

Given

dV	3290	m/s	
Mass of Habitat Unit	30000	kg	
Mass of Cryo Unit	11000	kg	
Mass of Payload	289417.0399	kg	
Max G (F/W)	6	earth G's	
Thrust to Weight Ratio	1.2		
g0	9.81	m/s^2	

Thrust Requirement-top level analysis

ISP	435	sec	
finert	0.24		
mass prop	637368.6101	kg	add 20% for residual
max inert mass	201274.2979	kg	
max initial mass	1128059.948	kg	
F	13279521.71	N	

Part 1: system mass and envelope

Me	13700.55093	kg
Le	731.6630503	cm
De	494.5639171	cm

Part 2: propellant
Oxygen and Hydrogen

O/F	3.8	
ISP	435	sec

Part 3 & 4:Engine cycle and cooling approach
Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000	Pa	selected	
Chamber pressure (Pc)	11500000	Pa	selected	
Atmospheric pressure(Pa)	650	Pa	vacuum	
γ	1.23		O/F = 3.8	
Tc	2800	K	O/F = 3.8	
MW	9.8	kg/kmol	O/F = 3.8	
c*	2402.69663	m/s	1.02	c* eff.

Output

Me	4.627484385	eq 3.100
ε	52.52696338	to get Pe
mdot propellant	3111.889511	eq 1.7
At	0.650167517	eq 3.133
Ae	34.15132535	using ε
mdot ox	2463.579197	O/F = 3.8
mdot fuel	648.3103149	O/F = 3.8

Determination of Propellant Needed at Mars

Mpayload	41000	kg
Minert	194560.3	kg
Isp	435	sec
g0	9.81	m/s^2
Delta V	3067	m/s
f inert	0.44	selected
M prop	248417	kg
f inert	0.43921	calculated

Estimate Masses and size tanks

Isp	435	sec
f inert	0.24	
payload	289417	kg
dV	3290	m/s

Estimate Masses

20% added for blow down extra

Mprop	637368.6	eq. 1.27	
Minert	201274.3	eq. 1.24	
Mfinal	490691.3		
Minitial	1128060		
Mfuel	132785.1	O/F = 3.8	
Mox	504583.5	O/F = 3.8	
Vfuel	1870.213		71
Vox	441.8419		1142

Size Tanks

	spherical tanks Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	504583.483	132785.1271
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.23E+09	1.23E+09
Tank Density (kg/m^3)	4460	4460
Vol. (5% Ullage)	463.9340255	1963.723711
radius (m)	4.802372816	7.768382327
area (m^2)	289.8154995	758.3523679
wall thickness (m)	0.007808736	0.012631516
mass tank (kg)	10093.39392	42722.96463
tank factor	6350	6350
mass tank using TF	29790.20447	126094.9352

Thrust Chamber			**Nozzle**		
N2O4/RP-1					
Input			**Input**		
Pc (Pa)	11500000				
c*	2402.69663		Nozzle Throat Dia	0.909846	
gamma	1.23		Ae	34.15133	
At	0.650167517		Nozzle exit Diam.	6.59415	
Ae	34.15132535		Cone 1/2 angle (deg)	20	
L* Table 5.6	0.5		Lf for eff=.985 fig 5.25	0.6	
Mc	0.4		thick. throat wall (m)	0.038596	
Ftu	3.10E+08	columbium	Using 1/2 chamber wall		
mult. factor	3		thick. nozzle exit	0.019298	
cham cont angle	45		density nozzle mater.	8500	columbium
dens comb cham	8500	columbium			
expansion ratio	52.52696338				
Output			**Output**		
Ac (m^2)	1.511418987		Ln (m) 15 deg nozzle	7.808749	
Chamber Len (m)	0.215085136		Ln bell nozzle (m)	4.68525	
Chamber Dia (m)	1.387226882		theta n	38	
Troat Dia (m)	0.909845588		theta e	13	
Len:Dia ratio	0.155046834				
Cham Thick (m)	0.077192464		*Using a non-tapered bell nozzle*		
Mass Cham (kg)	963.9118645		mass nozzle (kg)	18117.95	
			Using a tapered nozzle		
			f1	-0.004119	
			f2	0.606617	
			mass nozzle (kg)	12444.76	

Mass Summary

Total engine mass (using 50% for chamber & nozzle)	26817.34486
Mass injectors (15%)	6704.336216
mass oxidizer tank	29790.20447
mass fuel tank	126094.9352
support structure (10% of tank & engine masses)	18270.24845
feed system (system level est.-engine mass)	-13116.79393
Total Inert mass	194560.2752
propellant mass	637368.6101
payload mass	289417.0399
Final mass	483977.3151
Inert mass fraction	0.233866474
Initial mass of vehicle	1121345.925
Thrust	13279521.71
F/W	1.207184961
Check G limit	2.796973939

Lee Gentile
Mars mission design
Cryogenic Storage with T1000G tanks without IMPRESS

Given

dV	6357	m/s
Mass of Habitat Unit	30000	kg
Mass of Cryo Unit	11000	kg
Mass of Payload	41000	kg
Max G (F/W)	6	earth G's
Thrust to Weight Ratio	1.2	
g0	9.81	m/s^2

Thrust Requirement-top level analysis

ISP	435	sec	
finert	0.054		
mass prop	210274.5125	kg	add 20% for residual
max inert mass	12002.98486	kg	
max initial mass	263277.4974	kg	
F	3099302.699	N	

Part 1: system mass and envelope

Me	3811.939654	kg
Le	421.9807881	cm
De	254.4125507	cm

Part 2: propellant
Oxygen and Hydrogen

O/F	3.8	
ISP	435	sec

Part 3 & 4:Engine cycle and cooling approach
Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000	Pa	selected
Chamber pressure (Pc)	11500000	Pa	selected
Atmospheric pressure(Pa)	650	Pa	vacuum
γ	1.23		O/F = 3.8
Tc	2800	K	O/F = 3.8
MW	9.8	kg/kmol	O/F = 3.8
c*	2402.69663	m/s	1.02 c* eff.

Output

Me	4.627484385	eq 3.100
ε	52.52696338	to get Pe
mdot propellant	726.2827515	eq 1.7
At	0.151742358	eq 3.133
Ae	7.970565295	using ε
mdot ox	574.9738449	O/F = 3.8
mdot fuel	151.3089066	O/F = 3.8

Estimate Masses and size tanks

Isp	435	sec
f inert	0.054	
payload	41000	kg
dV	6357	m/s

Estimate Masses

Mprop	210274.5	eq. 1.27		20% added for blow down extra
Minert	12002.98	eq. 1.24		
Mfinal	53002.98			
Minitial	263277.5			
Mfuel	43807.19	O/F = 3.8		
Mox	166467.3	O/F = 3.8		
Vfuel	617.0027		71	
Vox	145.7682		1142	

Size Tanks

	spherical tanks Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	166467.3224	43807.19011
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.16E+09	1.16E+09
Tank Density (kg/m^3)	1550	1550
Vol. (5% Ullage)	153.056645	647.8528115
radius (m)	3.318342378	5.367794894
area (m^2)	138.3732848	362.0776265
wall thickness (m)	0.00572128	0.009254819
mass tank (kg)	1227.092067	5193.992368
tank factor	50800	50800
mass tank using TF	1228.512164	5200.003303

Thrust Chamber
N2O4/RP-1

Input

Pc (Pa)	11500000	
c*	2402.69663	
gamma	1.23	
At	0.151742358	
Ae	7.970565295	
L* Table 5.6	0.5	
Mc	0.4	
Ftu	3.10E+08	columbium
mult. factor	3	
cham cont angle	45	
dens comb cham	8500	columbium
expansion ratio	52.52696338	

Output

Ac (m^2)	0.352749523
Chamber Len (m)	0.215085136
Chamber Dia (m)	0.670175083
Troat Dia (m)	0.439550192
Len:Dia ratio	0.320938724
Cham Thick (m)	0.037292001
Mass Cham (kg)	182.8794854

Nozzle

Input

Nozzle Throat Dia	0.43955	
Ae	7.970565	
Nozzle exit Diam.	3.185661	
Cone 1/2 angle (deg)	20	
Lf for eff=.985 fig 5.25	0.6	
thick. throat wall (m)	0.018646	
Using 1/2 chamber wall		
thick. nozzle exit	0.009323	
density nozzle mater.	8500	columbium

Output

Ln (m) 15 deg nozzle	3.772439
Ln bell nozzle (m)	2.263464
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	2042.826

Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	1403.165

Mass Summary

Total engine mass (using 50% for chamber & nozzle)	3172.089902
Mass injectors (15%)	793.0224755
mass oxidizer tank	1228.512164
mass fuel tank	5200.003303
support structure (10% of tank & engine masses)	960.0605369
feed system (system level est.-engine mass)	639.8497522
Total Inert mass	11993.53813
propellant mass	210274.5125
payload mass	41000
Final mass	52993.53813
Inert mass fraction	0.053959794
Initial mass of vehicle	263268.0507
Thrust	3099302.699
F/W	1.200043059
Check G limit	5.961726807

Lee Gentile
Mars mission design
Cryogenic Storage with Titanium Tanks without IMPRESS

Given

dV	6357	m/s	
Mass of Habitat Unit	30000	kg	
Mass of Cryo Unit	11000	kg	
Mass of Payload	41000	kg	
Max G (F/W)	6	earth G's	
Thrust to Weight Ratio	1.2		
g0	9.81	m/s^2	

Thrust Requirement-top level analysis

ISP	435	sec	
finert	0.225		
mass prop	66511024.08	kg	add 20% for residual
max inert mass	19309652.15	kg	
max initial mass	85861676.23	kg	
F	1010763653	N	

Part 1: system mass and envelope

Me	704664.2419	kg
Le	31075.13031	cm
De	24025.21456	cm

Part 2: propellant
Oxygen and Hydrogen

O/F	3.8	
ISP	435	sec

Part 3 & 4:Engine cycle and cooling approach
Expander cycle
Regenerative cooling

Part 5: Pressure levels

Exit pressure (Pe)	15000	Pa	selected	
Chamber pressure (Pc)	11500000	Pa	selected	
Atmospheric pressure(Pa)	650	Pa	vacuum	
γ	1.23		O/F = 3.8	
Tc	2800	K	O/F = 3.8	
MW	9.8	kg/kmol	O/F = 3.8	
c*	2402.69663	m/s	1.02	c* eff.

Output

Me	4.627484385	eq 3.100
ε	52.52696338	to get Pe
mdot propellant	236859.7965	eq 1.7
At	49.48715086	eq 3.133
Ae	2599.409761	using ε
mdot ox	187514.0056	O/F = 3.8
mdot fuel	49345.79094	O/F = 3.8

Estimate Masses and size tanks

Isp	435	sec
f inert	0.225	
payload	41000	kg
dV	6357	m/s

Estimate Masses

20% added for blow down extra

Mprop	66511024	eq. 1.27	
Minert	19309652	eq. 1.24	
Mfinal	19350652		
Minitial	85861676		
Mfuel	13856463	O/F = 3.8	
Mox	52654561	O/F = 3.8	
Vfuel	195161.5		71
Vox	46107.32		1142

Size Tanks

	spherical tanks Composite Ox	Composite Fuel
Density Prop.	1142	71
Mass Prop/Press.	52654560.73	13856463.35
MEOP tank (Pa)	2000000	2000000
Burst Press. (Pa)	4000000	4000000
Ftu (Pa)	1.23E+09	1.23E+09
Tank Density (kg/m^3)	4460	4460
Vol. (5% Ullage)	48412.68719	204919.5284
radius (m)	22.60946018	36.57336437
area (m^2)	6423.773961	16808.91533
wall thickness (m)	0.03676335	0.059468885
mass tank (kg)	1053271.146	4458249.252
tank factor	6350	6350
mass tank using TF	3108683.069	13158324.92

Thrust Chamber

N2O4/RP-1

Input

Pc (Pa)	11500000	
c*	2402.69663	
gamma	1.23	
At	49.48715086	
Ae	2599.409761	
L* Table 5.6	0.5	
Mc	0.4	
Ftu	3.10E+08	columbium
mult. factor	3	
cham cont angle	45	
dens comb cham	8500	columbium
expansion ratio	52.52696338	

Output

Ac (m^2)	115.0408433
Chamber Len (m)	0.215085136
Chamber Dia (m)	12.1026671
Troat Dia (m)	7.937820698
Len:Dia ratio	0.017771714
Cham Thick (m)	0.673454863
Mass Cham (kg)	278483.2681

Nozzle

Input

Nozzle Throat Dia	7.937821	
Ae	2599.41	
Nozzle exit Diam.	57.52974	
Cone 1/2 angle (deg)	20	
Lf for eff=.985 fig 5.25	0.6	
thick. throat wall (m)	0.336727	
Using 1/2 chamber wall		
thick. nozzle exit	0.168364	
density nozzle mater.	8500	columbium

Output

Ln (m) 15 deg nozzle	68.12634
Ln bell nozzle (m)	40.87581
theta n	38
theta e	13

Using a non-tapered bell nozzle

mass nozzle (kg)	12031224

Using a tapered nozzle

f1	-0.004119
f2	0.606617
mass nozzle (kg)	8263943

Mass Summary

Total engine mass (using 50% for chamber & nozzle)	17084852.8
Mass injectors (15%)	4271213.199
mass oxidizer tank	3108683.069
mass fuel tank	13158324.92
support structure (10% of tank & engine masses)	3335186.079
feed system (system level est.-engine mass)	-16380188.55
Total Inert mass	24578071.51
propellant mass	66511024.08
payload mass	41000
Final mass	24619071.51
Inert mass fraction	0.26982452
Initial mass of vehicle	91130095.59
Thrust	1010763653
F/W	1.130625517
Check G limit	4.185129867